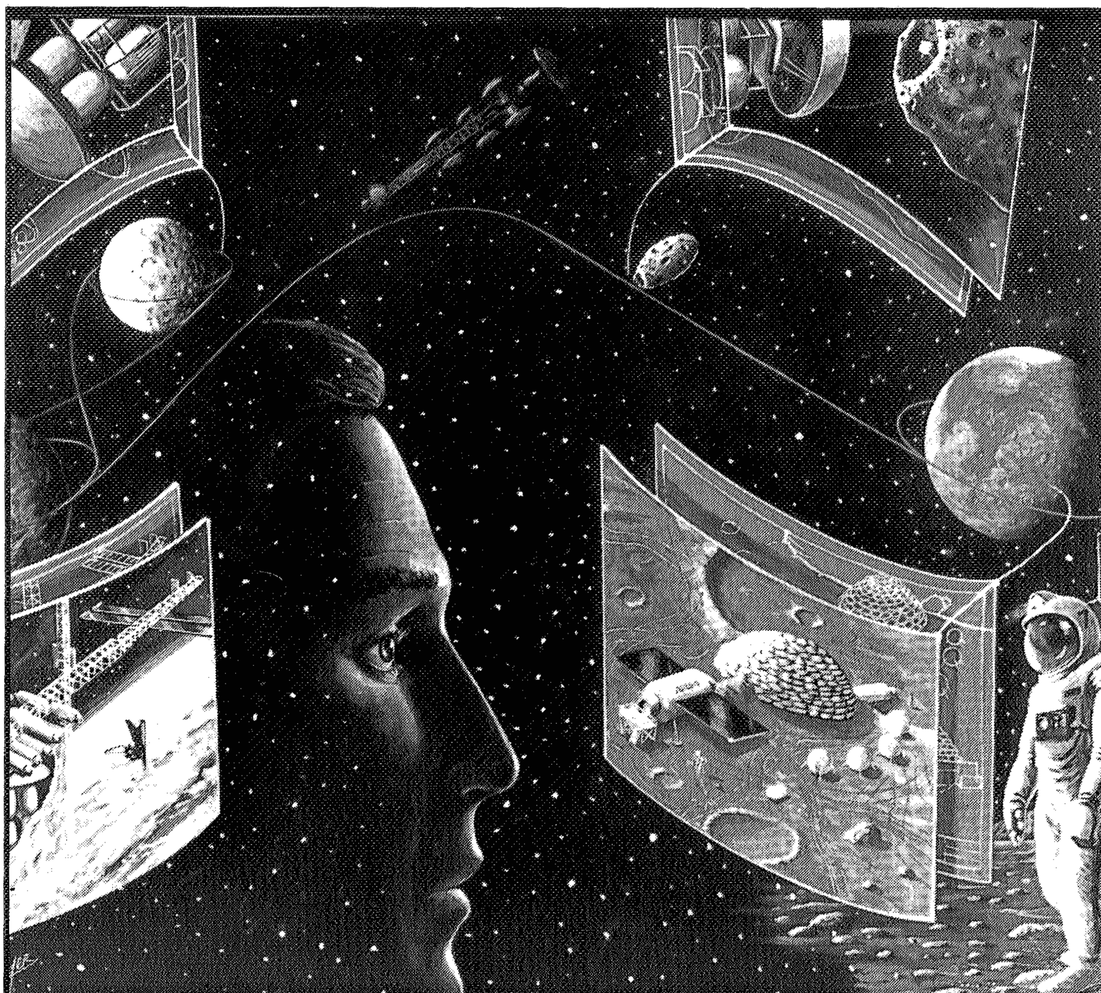
Technical Memorandum 4170

THE OFFICE OF EXPLORATION
FY 1989 ANNUAL REPORT



EXPLORATION STUDIES TECHNICAL REPORT
Volume I: Mission and Integrated Systems

NASA
National Aeronautics and
Space Administration

This publication is one of seven documents describing work performed in fiscal year 1989 under the auspices of the Office of Exploration. Volume 0, titled "Journey Into Tomorrow," provides an overall programmatic view of the goals, opportunities, and challenges of achieving a national goal for human exploration. The technical details and analyses are described in the other six volumes of the series. Volume I is Mission and Integrated Systems; Volume II is Space Transportation Systems; Volume III is Planetary Surface Systems; Volume IV is Nodes and Space Station Freedom Accommodations; Volume V is Technology Assessment; and Volume VI is Special Reports, Studies, and In-Depth Systems Assessment. These seven volumes document the status of Exploration Technical Studies at the conclusion of the FY 1989 study process in August 1989, and, therefore, do not contain any analyses, data, or results from the NASA 90-Day Study on Human Exploration of the Moon and Mars.

NASA Technical Memorandum 4170

The Office of Exploration FY 1989 Annual Report

Exploration Studies Technical Report Volume I: Mission and Integrated Systems

NASA

Disclaimer Statement

The Exploration Studies Process, as explained in detail in Section 2 of Volume I, was a requirements driven, iterative, and dynamic process developed for case study analysis. This process consisted of three parts: (1) requirements generation, (2) implementation development, and (3) integrated case study synthesis.

During the final step of the process, an integrated mission was developed for each of the case studies by synthesizing the implementations developed earlier into a coherent and consistent reference mission. These are presented in Section 3 of Volume I of this annual report. Given the iterative and dynamic nature of this process, there are two important items to note:

- **The integrated case studies do not always reflect a mission that has a direct one-to-one correspondence to the requirements specified in the March 3, 1989, *Study Requirements Document*. Many changes were made to these requirements prior to and during the synthesis activities when warranted.**

- **The integrated case studies presented in Volume I represent the results of the synthesis process. Volumes II, III, and IV are the implementation databases from which the integrated case studies were derived. Therefore, the implementations outlined in Volumes II, III, and IV are generally reflected in the integrated case studies, but, in some cases, the implementations were changed in order to be effectively included in the integrated case studies. These modifications are only briefly discussed in Volumes II, III, and IV.**

CONTENTS

Section		Page

Section		Page

Section		Page

TABLES

FIGURES

Figure | Page

Figure		Page

Figure		Page

SECTION 1

EXECUTIVE SUMMARY

For the past 2 years, the NASA Headquarters Office of Exploration (OEXP) has been leading a NASA-wide effort to provide recommendations and alternatives for a national decision on a focused program of human exploration of the solar system. The development and analysis of human exploration case studies provide the framework for the study process, which pulls together the expertise of all the NASA program offices, the NASA field centers and JPL, the scientific community, and industry.

The results of these studies are documented annually in the Exploration Studies Technical Report. The information reported herein reflects the status of exploration studies at the conclusion of the FY 1989 study process in August 1989. Therefore, no data or analyses developed to support the President's Human Exploration Initiative are discussed; that effort is described separately in the "Report of the 90-Day Study on Human Exploration of the Moon and Mars," published in November 1989. However, the studies documented in this volume did form the technical information base for both the national decision and NASA's 90-day study.

HUMAN EXPLORATION CASE STUDIES

The case study methodology provides the technical framework within which various approaches to human exploration of the solar system can be examined, analyzed, and compared. Within certain ground rules specified at the beginning of the study year, an end-to-end analysis and subsequent synthesis are performed, based on requirements for the elements that compose a human exploration approach: mission definition and architecture, science opportunities and strategies, transportation systems, orbital nodes, and planetary surface systems. The results of this process provide an understanding of the feasibility and benefits of the various case studies, enabling the identification of primary candidates for the most suitable approaches to the case study elements.

In FY 1989, three case studies were formulated for detailed development and analysis: Lunar Evolution, Mars Evolution, and Mars Expedition. The Lunar and Mars Evolution case studies are separate Moon-only and Mars-only approaches to a phased buildup and sustained outpost development strategy. The Mars Expedition case study, however, examined an alternative approach to the human exploration of Mars: a single mission program with the objective of embarking upon the earliest possible human landing on Mars.

The Lunar and Mars Evolution case studies were used to analyze and synthesize the development of permanent human outposts based on a strategy using multiple missions following three progressive phases: emplacement, consolidation, and utilization.

The emplacement phase emphasizes accommodating basic human habitation needs, establishing surface equipment and science instruments, and laying the foundation for future, more complex instrument networks and surface operations by testing prototypes of later systems. In the process, human explorers start learning to live and work on another planetary body, conducting local geologic investigations, performing experiments in mining the lunar soil to demonstrate the feasibility of oxygen production on the Moon, and examining the possibility of oxygen and water extraction on Mars. By the end of the emplacement phase, the support facilities include landing vehicle servicing equipment needed to prepare for longer visits. During this phase, human operations take place within tens of kilometers of the outpost, and unmanned rovers are used to explore more distant areas.

The consolidation phase further extends human presence, both in complexity of operations and distances traveled from the outpost, and continues to develop experience in living and working in an extraterrestrial environment. During this phase, outpost capabilities and scientific facilities are improved, more sophisticated instruments are installed, and power and pressurized volume are increased. A constructible habitat is erected at the outpost to provide the increased volume required for both extended crew residence and laboratory sciences research. Human operations expand to a range of hundreds of kilometers from the outpost.

Learning to become more independent of Earth now takes on paramount importance. This involves developing confidence in operational strategies as well as developing improved outpost element subsystems. More efficient systems for life support are emplaced, prototypes of resource processing plants are tested, and day-to-day activities are conducted without continual supervision and guidance from support staff on Earth.

The objectives of the utilization phase are to make routine use of in situ resources, and to continue to live and work at the outpost with minimal dependence on Earth. The area of exploration opportunities is expanded to include routine human access to more distant points on the planet.

The Mars Expedition case study examined a different approach to the human exploration of Mars, emphasizing the earliest possible date for landing on the surface. The case study uses an expendable vehicle strategy in which a single vehicle, with all supporting systems and the Mars landing/ascent vehicle attached, is launched intact into low-Earth orbit. Several subsequent launches bring up the trans-Earth

and trans-Mars propulsion stages. Following low-Earth orbit mating of the propulsion stages with the piloted vehicle, the three-member crew is launched to Mars in a zero-gravity vehicle on an opposition-class trajectory with a free flyby abort capability. The vehicle uses aerobraking at Mars and remains in Mars orbit for 30 days. A Mars lander, with all three crewmembers, descends to the surface for 20 days. Limited science equipment is carried to the surface of Mars. After their tour of duty is completed, the crew returns to Mars orbit for direct entry to Earth's surface. The nominal mission duration is approximately 18 months.

None of the focused case studies should be considered a proposal for implementation of a national exploration strategy. Rather, it was the goal of the FY 1989 exploration studies to develop the database from which subsets, or individual pieces, from the different case studies could support a defined approach. Furthermore, certain basic components of an exploration initiative are independent of the target or mission sequence, such as the need for a heavy lift launch vehicle, the need for expanded on-orbit operations, and life sciences issues, to name a few. A second goal of the case studies approach is to begin a dialogue in these particular arenas, and stimulate studies to bring an understanding, independent of mission, to a level that will support a national decision.

For each of the three case studies, analyses were conducted and syntheses performed for supporting infrastructure requirements, preparatory programs, and options, alternatives, and trades.

SUPPORTING INFRASTRUCTURE DESCRIPTION

All human exploration missions will originate, terminate, and be supported on Earth and/or in Earth orbit. In addition to the technological and scientific capabilities needed to expand human presence beyond Earth orbit, the human exploration program will require a supporting infrastructure to provide capabilities and services for establishing and maintaining a functional link to Earth. Significant compatibility and synergism between human exploration program plans and those of the other NASA programs that provide the supporting infrastructure are essential.

The exploration program supporting infrastructure will provide the required Earth-to-orbit and Earth-orbital transportation services, permanently manned Earth orbit facilities and services, and TNIM (telecommunications, navigation, and information management) support services as appropriate.

Earth-to-Orbit Transportation

Several Earth-to-orbit launch vehicle concepts have been identified to support the

FY 1989 case study requirements. These concepts range from current systems (Space Shuttle, expendable launch vehicles), through derivations of current systems (Shuttle-derived vehicles), to new classes of launch vehicles (e.g., the Advanced Launch System). For purposes of case study analysis, a generic concept was developed for a heavy lift launch vehicle with the capability to lift at least 140 metric tons to low Earth orbit. Although study results indicated that this concept was appropriate for the Mars Evolution and Mars Expedition case studies, the Lunar Evolution case study would best be accomplished using a mixed fleet of Shuttle and Shuttle-derived vehicles. Launch site ground processing improvements to support a new heavy lift launch vehicle were also identified.

Space Station Freedom

A detailed assessment was made of the implications on Space Station Freedom of supporting the case studies and implementation plans required to accommodate various space transfer vehicles, orbital support equipment, payloads, and additional support infrastructure. This implementation includes an engineering description of Freedom's systems used to meet the mission requirements, including habitat modules, laboratory modules, and vehicle processing facilities. Preliminary analysis shows that, as configured as of January 1989, the Space Station Freedom program can support all case study requirements by evolving the current Freedom configuration to a transportation node, provided all currently planned Space Station Freedom hooks and scars are maintained in the baseline program and that the appropriate heavy lift launch capability becomes available.

Telecommunications, Navigation, and Information Management

The telecommunications, navigation, and information management (TNIM) architectural and system design functions were defined for the case studies, and critical space- and Earth-based technologies that must be developed were identified. The TNIM architectures for the lunar and Mars case studies are similar in many ways, but substantial differences exist that translate directly into differing requirements. The distance between Earth and the Moon results in a one-way transmission delay of approximately 1.25 seconds, but from Mars, the delay can be as long as 20 minutes. In both cases, special approaches to conducting operations must be developed for human exploration and habitation of the Moon and Mars. Solutions focus on greater transmitter power, larger antenna dimensions, data compression, and the development of nearly autonomous systems.

PREPARATORY PROGRAM DESCRIPTIONS

Part of the development and analysis of a range of alternatives for expanding human presence in the solar system beyond low Earth orbit is the integration of capabilities provided by other NASA programs as they relate to the prerequisites for implementing one or more exploration alternatives. An assessment was made of the current NASA programs in life sciences, robotic missions, and technology development to determine the program capabilities required to implement the case studies.

Life Sciences

Human exploration of the Moon and Mars presents unique challenges for the life sciences and life support systems well beyond those required for the Space Shuttle or Space Station Freedom, and life sciences programs significantly impact the ability of the United States to undertake human exploration missions. The biological effects on humans resulting from extended exposure to space radiation, zero and/or artificial gravity, and isolated and confined environments must be understood.

To conduct the human exploration case studies, development is required in five critical life sciences areas. The first is advanced medical care to provide for remote care in the event of illness or injury, to develop methodologies for crew health maintenance and monitoring, and to identify requirements for crew medical skills. The second area, reduced gravity countermeasures, addresses the development of methods to maintain the health and physical capabilities of crews during exposure to zero, reduced, or artificial gravity and also to facilitate readaptation to Earth's gravity. The third area concerns the space radiation hazard to human explorers. It is important to determine the levels of risk of chronic low-dose, solar flare, galactic cosmic, and manmade radiation, and to develop appropriate countermeasures and warning capabilities.

In the fourth area, life support, processes need to be developed for space transfer vehicles, planetary surface outposts, EVA space suits, and planetary roving vehicles to revitalize air and water, supply food, and monitor and decontaminate the environment. In addition, protection and warning requirements associated with other hazards, such as toxic gases and micrometeoroids, must also be developed. Finally, the fifth area, space human factors, addresses the optimization of systems design requirements and measures to ensure safe, productive, and enhanced crew performance.

Robotic Missions

Robotic planetary missions serve as precursors to human exploration missions, determining the planetary environments in which spacecraft and crew must function, selecting an appropriate landing site, focusing the development of required technologies, and demonstrating the engineering capabilities needed to conduct human exploration of the Moon and Mars.

Because previous lunar exploration has provided an extensive base of information about the Moon, the Lunar Observer mapping mission appears to provide the appropriate data for the Lunar Evolution case study. For the Mars case studies, the 1992 Mars Observer and a future Mars Rover/Sample Return are the highest priority missions.

Technology Development

The case study methodology has been structured specifically to identify technologies that enable or enhance human exploration of the Moon and Mars. Special emphasis was placed on identifying technologies that are required to implement such missions or that would significantly increase the capability or decrease the cost of a particular mission. Critical technological requirements fall into three categories: near-term enabling, mid-term enabling, and far-term enhancing. Near-term enabling technologies are those that must be available prior to 2004 and are essential to mission success, from the standpoint of either technical feasibility and performance or affordability. Technologies in this category are required for both Lunar and Mars case studies. Those classed as "mid-term enabling" meet the same criteria, but they are needed in the time frame 2004 to 2010. Far-term enhancing technologies are those that, if available in the post-2010 time frame, would significantly benefit missions to Mars. Near-term enabling technologies include life support systems, EVA systems, radiation protection, cryogenic fluid management, cryogenic engines, efficient space transfer systems, and in-space vehicle operations. Mid-term enabling technologies include nuclear surface power, surface transportation, and in situ resource utilization. Far-term enhancing technologies are high-energy aerocapture, nuclear and solar electric propulsion, and gas-core nuclear thermal rockets.

OPTIONS, ALTERNATIVES, AND TRADES

In addition to the systems definition studies performed in direct response to specific focused case study requirements, the OEXP activities also include studies and product development in three categories. The first of these is special assessments, which focus on "high leverage" issues that are independent of specific case studies. The

assessments generally cover a broad subject area with potential for significant benefit to all mission options. Four special assessments — power systems, propulsion systems, life support systems, and automation and robotics — were conducted in FY 1989.

The second category is controlled trade studies, which affect more than one case study, are case-study independent, or involve multiple integration areas. The studies represent parametric analyses across a broad range of options, and the results are essential to further mature the case studies or enable a technical "assault" on the case study constraints. Three trades — Earth-Moon node location, lunar liquid oxygen leverage, and launch/on-orbit operations — were conducted in FY 1989.

The third category is emerging case studies, which are candidate focused case studies that are not yet sufficiently mature for release from the Mission Analysis and Systems Engineering analysis process. Upon further analysis by MASE and review by OEXP, an emerging case study may be adopted in part or total as a future focused case study, or it may be dropped from consideration. Two emerging case studies, the Lunar Oasis and a Near-Earth Asteroid Expedition, were developed.

Power System Special Assessment

Power system alternatives were reviewed for spacecraft and lunar and Mars surface applications. For spacecraft, the power options investigated were nuclear electric and solar electric for use in cargo vehicles. Studies showed that an SP-100-type thermoelectric reactor designed to power a Mars nuclear electric cargo vehicle provides savings in mass in LEO. Solar electric propulsion also showed potential for savings in mass as compared to chemical and chemical/aerobrake systems.

For surface systems, a concept was investigated for quick deployment of a nuclear power plant for use on the lunar surface. Studies indicated that the SP-100-type thermoelectric reactor integrated into a lander could provide sufficient power for the early stages of both lunar and Mars evolutionary growth requirements. Potential power systems for mobile power (rovers) were also examined.

Propulsion Systems Special Assessment

Several studies conducted during FY 1989 focused on comparing propulsion systems and quantifying the advantages of their use for transportation to Mars. Chemical rockets were compared with nuclear thermal rockets, both with and without aerobrakes, for the Mars Expedition and Evolution case studies. Several more advanced propulsion technologies were investigated to identify and compare their uses for a Mars cargo mission.

Both chemical with aerobraking and nuclear thermal rocket systems show a large

benefit (in some cases more than 50 percent) over all-chemical propulsion for all Mars transportation missions. Chemical/aerobrake and nuclear thermal rocket systems are competitive, with nuclear thermal rockets showing advantages for the Mars Expedition case, and chemical/aerobrake showing advantages for the Mars Evolution case for the conditions considered. Variables such as aerobrake efficiency (mass fraction) and trajectory/mission design strongly affect the leverage of propulsion systems. The use of nuclear thermal rockets with aerobraking has the highest leverage of systems included in this study for all missions considered.

Another study examined technologies judged to be beyond advanced chemical, aerobrake, nuclear thermal rockets, and nuclear and solar electric propulsion, but available within the 25- to 30-year time frame of interest needed for exploration mission planning. These included solar sails, very high-power nuclear and solar electric propulsion, solar and laser thermal propulsion, rail guns and mass drivers, and tethers. Of these, solar sails and very high power nuclear and solar electric propulsion warrant further study for broad application to exploration missions.

A third study was conducted to identify fuel systems architectures to support transportation vehicles for exploration missions and other requirements. Preliminary results obtained in FY 1989 are expected to have broad implications for other options, and will help reduce the number of architecture options that need to be investigated.

Life Support Systems Special Assessment

Life support systems were examined using the generic scenario of a long-term lunar outpost as the study baseline. Four different approaches were examined: (1) an open loop Shuttle-type approach with a regenerable carbon dioxide removal process, (2) a closed-loop regenerative physical-chemical system such as that planned for Space Station Freedom, (3) a hybrid physical-chemical/biological system that represents an advancement over the Space Station Freedom system, and (4) a biological system for nearly complete resource recycling and life support. Extravehicular systems and portable life support systems were also assessed.

The first approach, open-loop systems, has very large mass requirements. Regenerative approaches, either physical-chemical or hybrid physical-chemical/biological, offer significant resupply and logistics mass savings, and, therefore, would be appropriate for evolving outposts and Mars transfer vehicles, where initial mass to orbit and resupply are primary considerations. Of the two regenerative approaches, the Space Station Freedom system significantly reduces expendables associated with air and water supply but requires system logistics to maintain operation, and it may not be totally appropriate for lunar and Mars missions. Hybrid regenerative sys-

tems, on the other hand, tend to have a higher initial mass, but require less resupply, depending on the degree of closure. Finally, in situ oxygen used for gas makeup in the habitat air supply does not offer as much advantage as in situ water, but in situ oxygen could be used with hydrogen to produce water. In situ water allows flexibility and can be used to expand an existing regenerative system.

Automation and Robotics/Human Performance Special Assessment

Special assessments in automation and robotics/human performance were directed at obtaining a better understanding of requirements in these areas for human lunar and Mars missions. In particular, analyses focused on barrier issues, critical problems, and high-leverage areas. Conventional and unconventional systems, technology, configurations, and technical options were evaluated, and high-level trade studies were performed. The goal of the assessment was to provide system analysis and design capability to enable effective allocation of functions between humans and machines for human exploration missions. The study was an iterative process that incorporated new information as it became available.

Overall, the assessment revealed that the current level of automation and robotics technology cannot handle the increasingly complex mission scenarios. Advances in automation and robotics technology are required to provide the semi-automated construction, maintenance, and repair capabilities that are critical to mission success. Likewise, advances in large-scale expert system technology are necessary to enable semiautonomous operation of the piloted spacecraft and planetary surface outposts.

Earth-Moon Node Location

The objective of this controlled trade study was to determine the most advantageous location for a transfer node for lunar missions, and to characterize this node location in terms of performance, functionality, and cost. Potential node locations in Earth-Moon space include low and geosynchronous Earth orbit and five libration points.

The performance and functionality assessments indicate that low lunar orbit and the L2 and L5 libration points are the best candidates for a node facility outside low Earth orbit. The direct-to-surface option also offers the interesting possibility of performing all assembly, fueling, and tests of the lunar vehicle in low Earth orbit for a modest performance penalty. Preliminary cost assessments support the contention that the use of lunar-derived oxygen can reduce overall cost.

Lunar Liquid Oxygen Leverage

The objective of this controlled trade study was to determine the lunar outpost activity level at which the production of oxygen from lunar materials becomes attractive as a means of reducing total program costs. By varying a number of parameters, such as number of crew and outpost activity level, it was possible to examine and analyze the impact of these parameters on oxygen demand. Study results indicate that using lunar liquid oxygen in the life support system, for lunar science missions, and for transportation between the lunar surface and low lunar orbit for an outpost with a crew of four results in significant total program cost savings within 10 years of operation.

Launch/On-Orbit Processing

The objectives of this controlled trade study were to analyze the trades associated with launching, assembling, and servicing a space transfer vehicle and to derive an effective blend of Earth-to-orbit launch capability, transfer vehicle design, and ground and space processing capabilities to support a given set of mission requirements. For FY 1989, a methodology and the proper analytical tools were developed for this trade study. In addition, the trade space of launch vehicle capabilities was assessed to provide information on required processing facilities, relative costs, and payload ground facilities. Also, the required on-orbit transfer vehicle processing facilities, equipment, and tasks were determined. Future work to complete the trade study will include assessing the proper blend of automation and robotics versus extravehicular activity, and then determining the relative cost of the on-orbit facilities and operations required to process space transfer vehicles.

Lunar Oasis Emerging Case Study

The Lunar Oasis emerging case study is an effort to define an alternative lunar outpost development strategy that considers self-sufficiency the major goal of the facility. The rationale for an effective architecture of the case study was developed to determine the magnitude of the space transportation logistics requirements, to understand new technology, and to decide what prerequisite science information should be obtained. It was concluded that the Oasis is a feasible approach, and elements of it should be studied in conjunction with future lunar case studies.

Near-Earth Asteroid Expedition Emerging Case Study

The principal reasons for considering the human exploration of near-Earth asteroids are to assess potential resources and to enhance scientific understanding of the history of the solar system. In addition, missions to asteroids could be carried out as an alternative to a Phobos mission, if a piloted mission to an asteroid-like body is re-

quired as part of the development of capability for later missions to Mars. Although near-Earth asteroids pass within the orbital distance of the Earth from the Sun, simple missions to these targets do not abound. No missions were discovered during this assessment that appeared to offer a near-term alternative to Phobos missions in terms of reducing round-trip times for reasonable initial masses in low-Earth orbit. It is, however, possible that new near-Earth asteroids with more accessible orbits will be discovered in the future.

SUMMARY

The goal of the FY 1989 Exploration Studies was to develop the database from which subsets of different case studies could support a defined approach. Several key findings, in areas including Mars trajectories, propulsion systems, vehicle design, planetary surface systems, orbital nodes, and launch vehicles, have emerged from these studies. The key findings of the 1989 studies combine with those of the 1988 studies to build the foundation of future analyses to determine how best to achieve the national goal of expanding human presence and activity beyond Earth's orbit into the solar system.

SECTION 2

Introduction

The NASA Office of Exploration (OEXP) was established in June 1987 to provide recommendations and alternatives for a national decision on a focused program of human exploration of the solar system, particularly of the Moon and Mars. OEXP is responsible for steering Agency investments on a practical, year-by-year basis toward providing feasible, defined choices in prerequisite programs and capabilities. With management centralized at NASA Headquarters, OEXP leads a NASA-wide team consisting of many of the program offices and field center organizations, where existing and future programs, plans, and support functions would be integral to the implementation of a national human exploration initiative.

2.1 EXPLORATION STUDIES

2.1.1 Study Objectives

FY 1989 is the second year of OEXP-sponsored exploration studies. Based on last year's experience, and on the developing and changing environment of the civil space program, a specific set of FY 1989 objectives was developed to guide the content, schedule, and milestones of the year's study plans.

The primary and continued goal is to develop a mature knowledge base to enable a decision on a program for human exploration. For the second year of studies, the objectives that were developed to achieve this goal are: (1) update and refine exploration cases for detailed study, (2) obtain a more detailed understanding of prerequisite requirements, (3) continue building the capabilities of the exploration study team, and (4) develop effective internal and external communications. The specific strategies to be employed this year to ensure the achievement of objectives 1, 2, and 3 are described below. The development of objective 4 was a NASA Headquarters OEXP function and is not reported in this document.

Update and Refine Exploration Cases. For FY 1988 exploration studies, the approach and methodology were actually formulated somewhat in parallel with the studies. Lessons learned and experience gained last year made it possible to begin this year's activities with a methodology already in place. Activities in FY 1989 were systematically conducted to ensure the determination of cause and effect of study results. Controlled trades and parametrics were particularly emphasized, so that causes and alternatives for major obstacles and special opportunities could be evaluated. In addition to refining last year's cases, new case studies were initiated.

Obtain Detailed Understanding of Prerequisite Requirements. The Office of Exploration continued to coordinate with the NASA Headquarters program offices to determine prerequisite requirements for Earth-to-orbit transportation, life sciences research, robotic scientific missions, Space Station Freedom utilization, telecommunications, navigation, information management, and technology. The strategy for FY 1989 focused on seeking to sort out truly enabling prerequisites from those that simply enhance. Emphasis was placed on exploiting the systems and infrastructures that will be in place in the late 1990s for initiating exploration. Areas more fully developed in FY 1989 include the initiation, with the Office of Space Science and Applications (OSSA), of science studies and user requirement and opportunity development. Also with OSSA, the OEXP continued to study artificial gravity research facility feasibility and concepts.

Continue Building Exploration Team Capability. The core exploration study team is in place and working effectively. The continued growth of this in-house systems engineering capability at the NASA field centers is of primary importance. In addition, it must be ensured that the maximum amount of innovation is extracted from external participants and injected into the study process.

2.1.2 Study Team

The NASA Exploration Study Team is presented in figure 2.1-1. The "Level" designation refers to activities and/or functions performed (as defined on the figure), not necessarily the geographic location of the personnel. Detailed descriptions of the Level I and Opportunity Assessment Agent activities and responsibilities are presented in the OEXP Exploration Management Plan. The Level II and III activities are defined below.

At Level II is the Mission Analysis and Systems Engineering (MASE) function. MASE is the top-level architect for case study development and definition, integrator of the focused case studies, and system engineer for the emerging case studies and controlled trade studies. MASE tasks include:

a. Case study development and definition
b. Requirements integration and results synthesis
c. Accommodation of science payloads and other opportunities as defined by the opportunities definition process
d. Incorporation of special assessment results
e. Cost understanding/methodology
f. In-space operations analysis

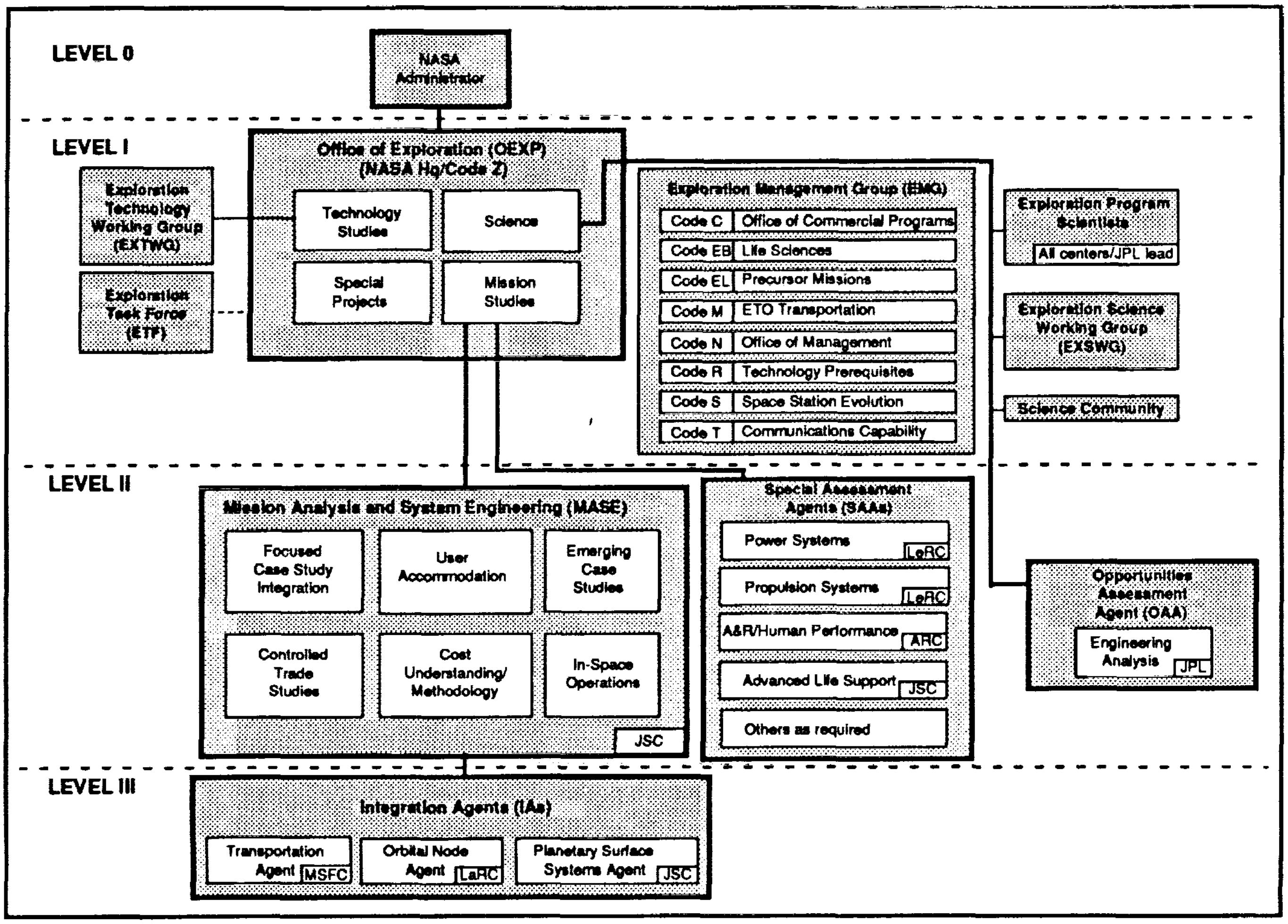


Figure 2.1-1.- NASA exploration study team.

Also at Level II are the Special Assessment Agents (SAAs), who perform discipline-related studies relevant to technological, operational, or programmatic needs as defined in the case study development process. The studies are related to Integration Agent activities (described below) in that a specific discipline area is identified in a study domain. The SAA focuses the discipline area to provide benefit, risk, and potential implementation details. The SAA is loosely coupled with the in-line day-to-day scenario development activity so that: (1) results from the SAA studies are periodically reviewed for potential incorporation into baseline scenario development activity, and (2) assignments to the SAA are periodically made based upon findings, problems, and/or issues arising from scenario development activity. Otherwise, the SAAs have significant latitude in pursuing their objectives. This year's special assessments were conducted in the areas of power, propulsion, life support, and automation and robotics. The results of these assessments are provided in sections 6.1, 6.2, 6.3, and 6.4 of this volume.

Level III consists of the Integration Agents (IAs), who perform system concept studies and trades and develop configurations, requirements, and implementation plans. These activities are functional assignments to NASA Centers that require the collection of inputs from one or more organizations that contribute subsets of the assignment or provide special assessment products. The IA is responsible for bringing together the team of contributing organizations within the assignment constraints imposed by OEXP. IA tasks and responsible organizations are provided in table 2.1-I.

2.1.3 Study Process

To accomplish OEXP objectives, a study process was developed that begins with the yearly articulation by OEXP of guidelines and ground rules for human exploration studies. This activity defines a framework of initial concepts within which alternative strategies can be formulated and explored.

The study process began with the release by Level I OEXP of the Exploration Requirements Document. Taking direction from Level I, a baseline Study Requirements Document (SRD) was prepared and released by MASE. This activity initiated the case study develop-

TABLE 2.1-I.- INTEGRATION AGENTS AND RESPONSIBLE ORGANIZATIONS

Space Transportation Systems — Marshall Space Flight Center (MSFC) a. Configuration definition and integration b. Integrated transfer vehicle configuration analysis c. Vehicle element definitions and trades d. Vehicle element descriptions
Planetary Surface Systems — Johnson Space Center (JSC) a. Requirements collection and synthesis b. Surface systems definition and analysis c. Surface transportation d. Surface systems descriptions
Nodes and Space Station Freedom Accommodation — Langley Research Center (LaRC) a. Node requirements collection and synthesis b. Node integrated analysis c. Assembly and turnaround integrated assessment d. Propellant depot definition/analysis support (LeRC) e. Node element descriptions

ment/validation phase, an iterative process based upon a series of internal trades involving MASE and the study agents. Closure on the case studies development was signalled by the release of a draft SRD, which included the preparation of a preliminary opportunities definition. The study agents then had 6 to 8 weeks for analysis based upon this draft SRD.

At Cycle 1 Review, all aspects of the program were reviewed, and any redirections to the study effort were made. Using inputs from the review, the final SRD was released. This activity initiated Cycle 2. Cycle 2 includes a more detailed analysis of focused case studies, including opportunity analysis and accommodation studies based upon the prior opportunities definition work. A Cycle 2 mid-term was scheduled to review progress. The final month of Cycle 2 was reserved for synthesis activities in preparation for Administrator briefings and final report writing.

Running concurrently with the Cycle 1 and 2 focused case study activities are the independent investigations and program support activities shown in figure 2.1-1. Although these activities were somewhat autonomous to the mainline studies, their results were reviewed and incorporated into the studies, if appropriate, at Cycle 1 Review.

FY 1989 Exploration Studies began in early November 1988 at a strategic planning workshop held in Williamsburg, Virginia, where white papers for proposed FY 1989 case studies were reviewed. Following this meeting, the MASE organization spent the next several weeks developing detailed requirements against which the agents were to develop their implementations. In parallel with the MASE requirement definition period, the IAs, SAAs, and NASA program offices began a top-level implementation activity, working within the white paper guidelines. Working Group Week 1 was held in December 1988 to review requirements and agent study plans and implementations to date.

A draft SRD was released by MASE on January 20, 1989, to the IAs, SAAs, and NASA program offices for comment and review. Working Group Week 2 was held at the end of February to finalize the study requirements for FY 1989. The baseline SRD was released in early March, signalling the beginning of Cycle 2. During Cycle 2, the exploration agents conducted detailed implementations within the guidelines specified by the SRD, with a mid-term review at Working Group Week 3 in late April. The exploration community's critique of the Agents' implementations at Working Group Week 3 led to several changes in the case studies and the requirements. This was an iterative procedure that continued through the end of the year; however, no formal update to the SRD was released. Therefore, the integrated reference missions for each focused study reported in this volume do not necessarily correspond one-to-one with the March SRD.

The exploration agents presented their final implementation plans to MASE on June 2, 1989. This initiated the MASE integration and synthesis activity, which continued until mid-July. During the synthesis activity, MASE integrated these implementation plans into an integrated reference mission for each focused case study. At Working Group Week 4, conducted July 12-14, MASE presented the synthesized focused case study results to the exploration study team for their critique. Based upon comments received at Working Group Week 4, MASE conducted another iteration in the synthesis process to arrive at the integrated missions reported in this volume. Following that, the annual report writing activity was initiated, culminating in the production of this volume.

2.2 CASE STUDY PROCESS

The examination of exploration options is executed through a case study process. The process develops top-level definitions of systems that implement one or more exploration strategies and accommodate the associated requirements within the framework of national space exploration goals. The process consists of iterative study phases, as illustrated by figure 2.2-1.

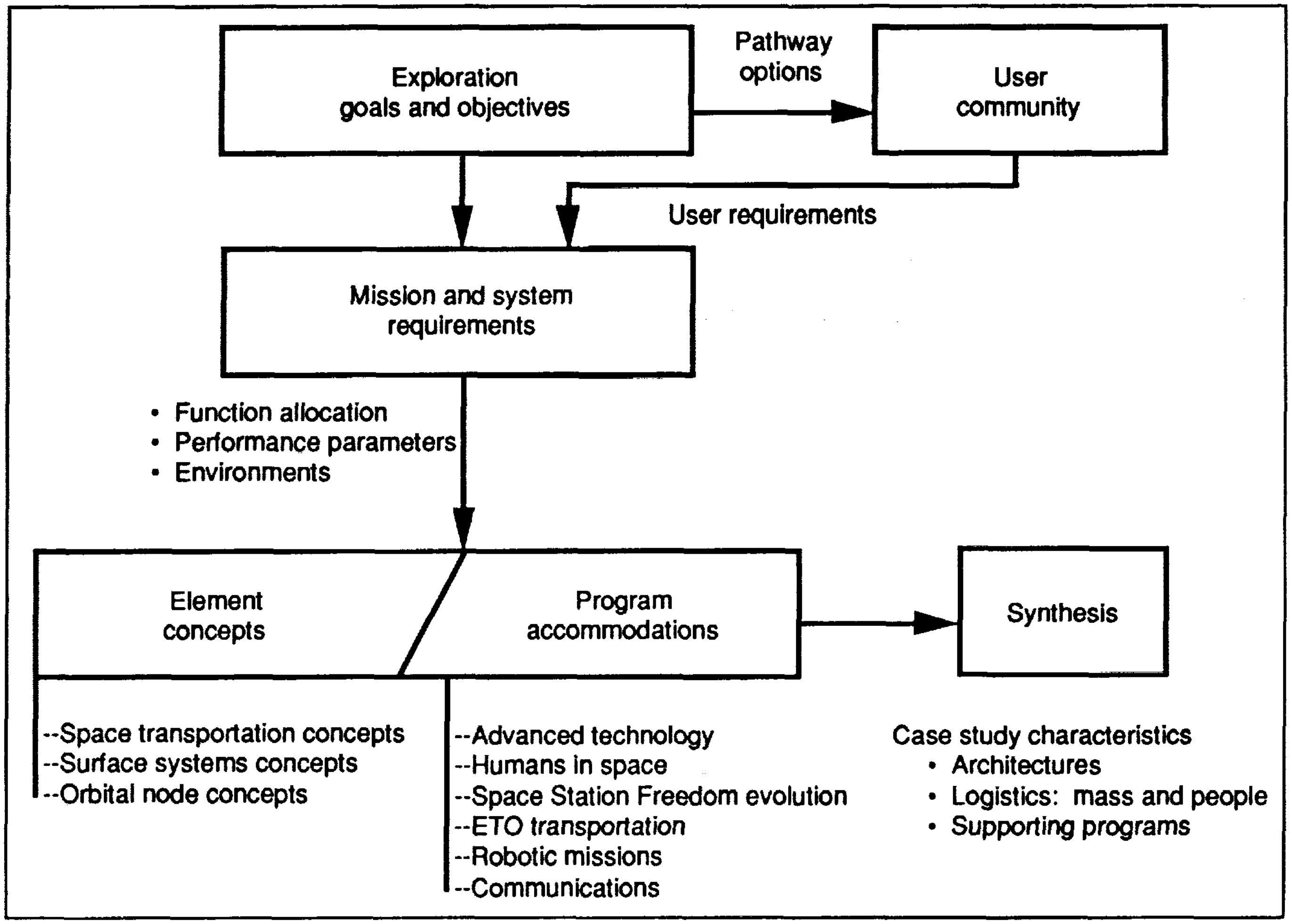


Figure 2.2-1.- Case study methodology.

The first phase addresses conceptual mission analysis and associated broad trades. During this period, mission requirements are defined that meet the exploration goals and objectives and user requirements. The mission requirements specify performance parameters for systems defined by this study, identify environments in which the conceptual systems must operate to meet the specified requirements, and point toward the broad trade areas. The technical options available within each trade area are analyzed for their relative benefit. These trades identify the system concept options and elements that outline the case study.

The second phase of the case study process, system engineering, encompasses system-level studies and syntheses using the results of the previous phase. Three domains of interest were identified as significant study areas: space transportation systems, planetary surface systems, and orbital nodes, all of which are, in general, programmatically independent and can be addressed initially as functionally independent. A definition of each system was incorporated into the mission-specific case studies.

The third phase comprises a synthesis of the system-level studies, in which system requirements assumptions provide a basis for defining configuration options, and system-level trade studies identify the parametric cost, performance, and risk. The results also establish a preliminary system concept and a reference configuration that is used to refine the study through several iterations. Where unique science and/or technology needs such as the implementation of nuclear spacecraft propulsion, were identified, special studies or assessments were made to identify strategies to accommodate those needs. The refined case studies, associated requirements, and relative benefits become the knowledge base of exploration path sensitivities. The base will be used to define the exploration initiative options, benefits, and risks that will support the selection and subsequent decision.

Figure 2.2-2 summarizes the FY 1989 case study development schedule.

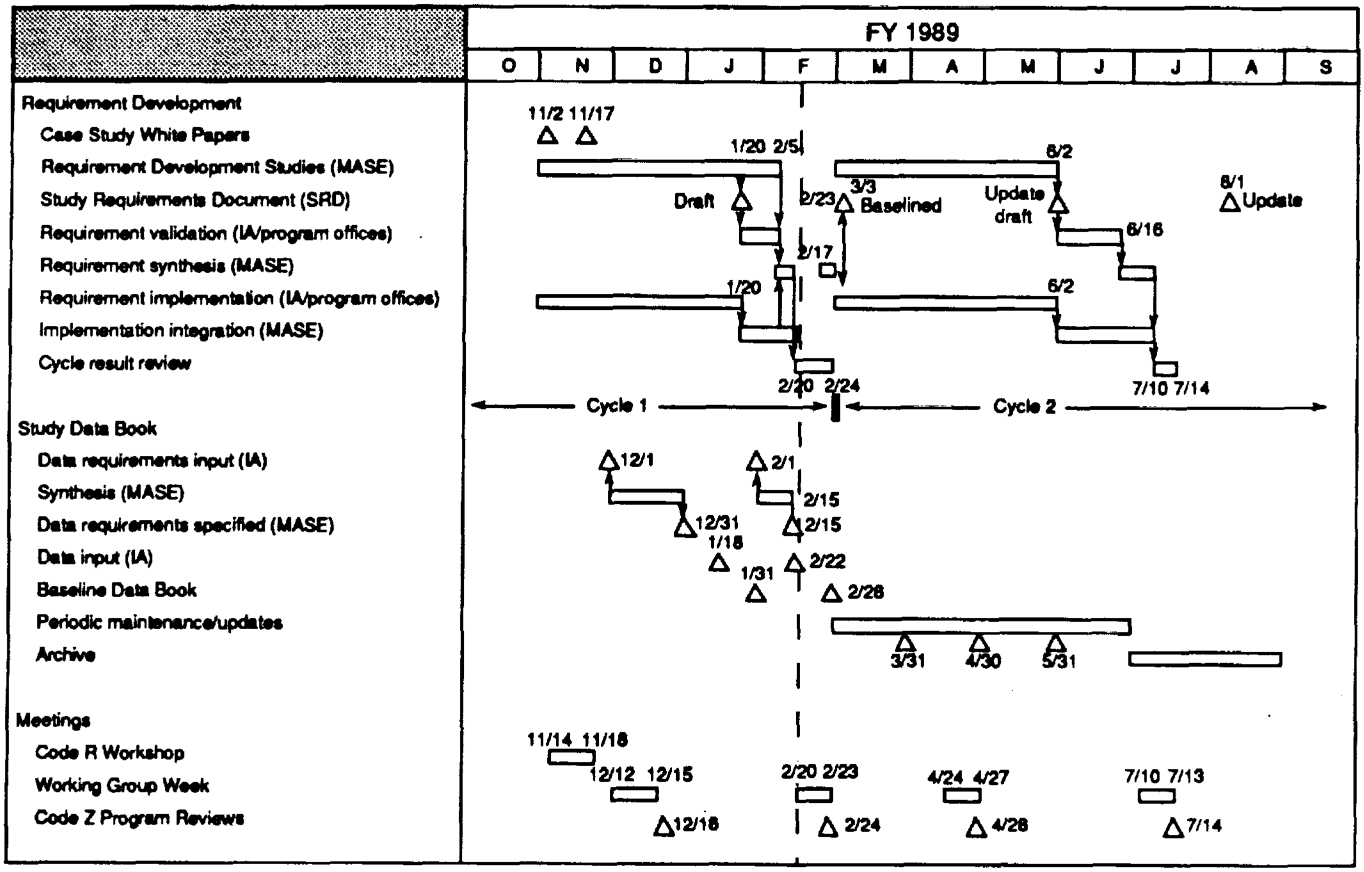


Figure 2.2-2.- Case study development schedule.

2.3 SCIENCE/USER COMMUNITY INTERACTION

At the request of OEXP, a team of NASA Center Exploration Program Scientists generated a draft report on Science Opportunities for Human Exploration. A second group of scientists, the Exploration Science Working Group (EXSWG), reviewed this report, and it was published as the Exploration Opportunities Document (EOD). The EOD was distributed to two science strategy working groups: one at Ames Research Center for Mars science, and the other at Johnson Space Center for lunar science. These working groups examined the science opportunities and molded them into coherent, integrated science strategies for lunar and Mars exploration options. These science programs were not limited by the MASE focused case study constraints, but were developed with consideration of the broad context of the Moon-to-Mars exploration objectives. Included in the science program development were site selections for exploration, sampling, study, and instrument deployment; sequencing of different parts of the scientific research; and a science rationale for the program; i.e., what research, experiment, or field work is done where, when, and why. Precursor activities were also discussed by the science strategy groups.

JPL's Opportunity Analysis Group works to provide engineering specifications and equipment and delivery options for the instrumentation constituting the science payloads for the integrated science program. The list of instruments that JPL characterizes is not limited to the science payloads evolving from the JSC/ARC-led efforts, and will contain a wider selection of science equipment that is most of, but not necessarily all, the instrumention proposed in the EOD.

Based on the engineering and technology opportunities of the mission scenarios, MASE then worked to accommodate the science payloads proposed by the JSC/ARC integrated science program and rationale. MASE partitioned the payloads into mission phases defined by the scenario, using the science strategy group's product to accommodate the highest priority research. MASE's overriding objective was to accommodate the optimal mix of high-priority human and unmanned scientific research options for each phase of the missions.

2.4 ANNUAL REPORTING

The study process culminates in the production of an annual report in seven volumes, as shown in figure 2.4-1. Volume 0, "Journey into Tomorrow," contains a

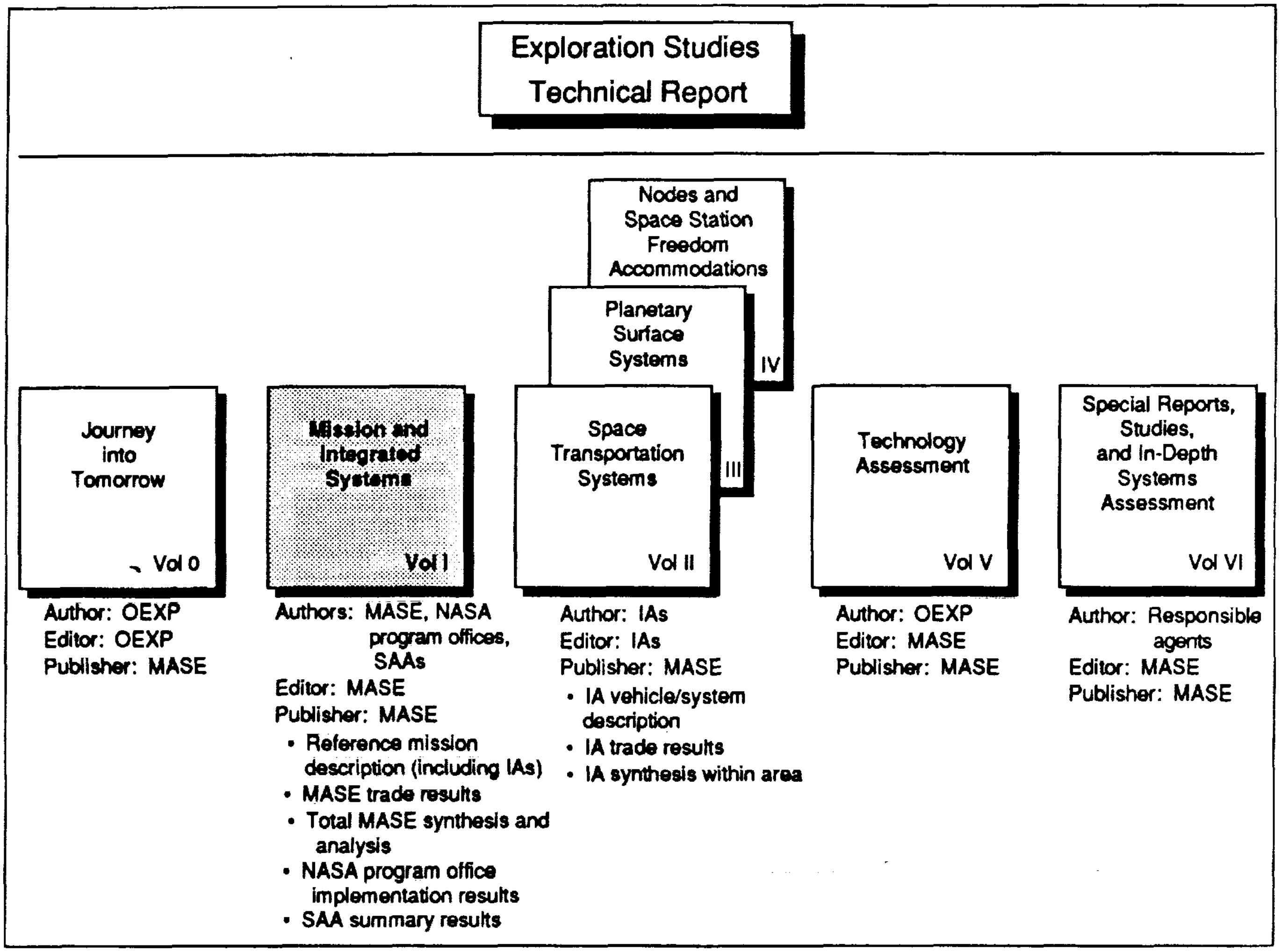


Figure 2.4-1.- FY 1989 Annual Report.

high-level technical and programmatic summary of the year's effort. Volumes I through VI document in detail the technical studies conducted in FY 1989.

The contents of Volume I, contained herein, are described below. Volumes II, III, and IV describe the vehicles and systems, controlled trades, and synthesis activities conducted by the Integration Agents in Space Transportation Systems, Planetary Surface Systems, and Nodes and Space Station Freedom Accommodations respectively. Volume V describes the year's technology assessment activity, and Volume VI is a collection of Special Reports, Studies, and In-Depth Systems Assessment.

This volume, Mission and Integrated Systems, contains the results of the total MASE synthesis and analysis for FY 1989. The detailed technical material begins with section 3, Human Exploration Case Studies, which summarizes the three FY 1989 case studies: Lunar Evolution, Mars Evolution, and Mars Expedition. For each case study, this section describes the reference mission and associated science opportunities and objectives, transportation systems, orbital node systems, and planetary surface systems. The reference mission descriptions include the synthesized results of the IA inputs. (Volumes II, III, IV, and V are the databases from which the Volume I integrated mission results are drawn, and these volumes, therefore, contain much more detail, along with important trade study results.)

The NASA Headquarters program office implementations are contained in sections 4 and 5, Supporting Infrastructure and Preparatory Programs respectively. Section 6 is a summary of the four SAA studies and assessments and the three internal MASE controlled trade studies. The analyses conducted by MASE for emerging case studies are also included in this section. Section 7 discusses overall study conclusions.

In the course of the detailed definition and assessment of the case studies, a significant level of understanding has been gained, regarding both specific case studies and human exploration missions in general. These topics, as well as other results of the FY 1989 study activity, are summarized in this volume.

SECTION 3

Human Exploration Case Studies

This section discusses the three focused case studies selected for investigation during FY 1989: (1) Lunar Evolution, (2) Mars Evolution, and (3) Mars Expedition. These three case studies are described in sections 3.1 through 3.3, which also outline the synthesized results prepared by the Mission Analysis and Systems Engineering (MASE) team, based on FY 1989 inputs and analyses from the Transportation, Orbital Node, and Planetary Surface System Integration Agents (IAs). Consideration was also given to the alternatives identified by the Special Assessment Agents (SAAs).

Sections 3.X.1 and 3.X.2 describe the key features and science opportunities and strategy for each case study. Sections 3.X.3, 3.X.4, and 3.X.5 detail the element concepts and configurations that fall within each IA's domain. These sections also include a description of the baseline program, any required technology advances identified within the area, and alternative approaches that were developed in the course of the investigation. The material in these sections reflects the baseline case studies defined in the Exploration Requirements Document (ERD) and the Study Requirements Document (SRD). Section 3.X.6 contains the synthesized mission manifest. Section 3.4 summarizes the case studies.

3.1 LUNAR EVOLUTION CASE STUDY

The objective of the Lunar Evolution case study is to establish a permanent, manned lunar outpost, which supports significant science objectives and serves as a test-bed and stepping-stone for further human exploration of the solar system. The primary emphasis is to gain experience in living and working on another planetary body for extended periods of time. Significant test programs are conducted on the Moon to validate the systems and operational concepts required for future exploration missions. In addition, the outpost develops a high measure of self-sufficiency, in terms of both logistical support and operations, to allow future exploration initiatives to be undertaken within a steady level of national investment.

3.1.1 Key Features

Key features of this case study include:

a. First piloted flight to the Moon in 2004.

b. Vehicles are expended for the first 2 years.

c. Vehicles are serviced at Space Station Freedom and on the lunar surface beginning 2 years after the first flights to the Moon.

d. Significant science research capability is provided.

e. Initial number of crew is four, increasing to 12 in 2012.

f. Aerobraking is employed on Earth return.

g. Chemical propulsion is utilized.

h. Propellant (lunar oxygen) production on the lunar surface begins in 2012.

i. Earth-Moon trajectories employ free abort.

3.1.1.1 Phases of Development

The lunar outpost is established through three evolutionary phases of development: (1) emplacement, (2) consolidation, and (3) utilization. These three phases are summarized in figure 3.1.1-1, and described below.

Emplacement Phase. The objective of this phase is to establish a permanent human presence on the Moon and begin developing a knowledge base of how to live and work on a nonterrestrial body. Emphasis focuses on emplacing equipment and instruments on the lunar surface and laying the foundation for later, more complex surface operations. Human operations are restricted to a local region about the outpost (tens of kms). The Earth-Moon transportation infrastructure is emplaced, and preparations are made to reuse the transportation vehicles and service them at Space Station Freedom and on the lunar surface.

Consolidation Phase. During this phase, human presence on the Moon increases, and operational experience on a nonterrestrial body continues to expand. Emphasis shifts to constructing surface systems, providing large pressurized volumes, setting up more complex scientific instruments and laboratories, and testing and developing systems and prototypes to be used in further planetary exploration. Human operations expand to a region within hundreds of kilometers of the outpost. Reducing the resupply from Earth is of paramount importance for both making resources available for further exploration and developing confidence in operational strategies and element subsystems needed for future exploration. This reduction is accomplished by (1) providing the lunar outpost with more efficient and reliable systems for life support and operations, and (2) testing prototype in situ resource production plants prior to their full implementation in the utilization phase. Lunar outpost independence is also enhanced by performing day-to-day activities in the absence of continual supervision and guidance from support staff on Earth. The Earth-Moon trans-

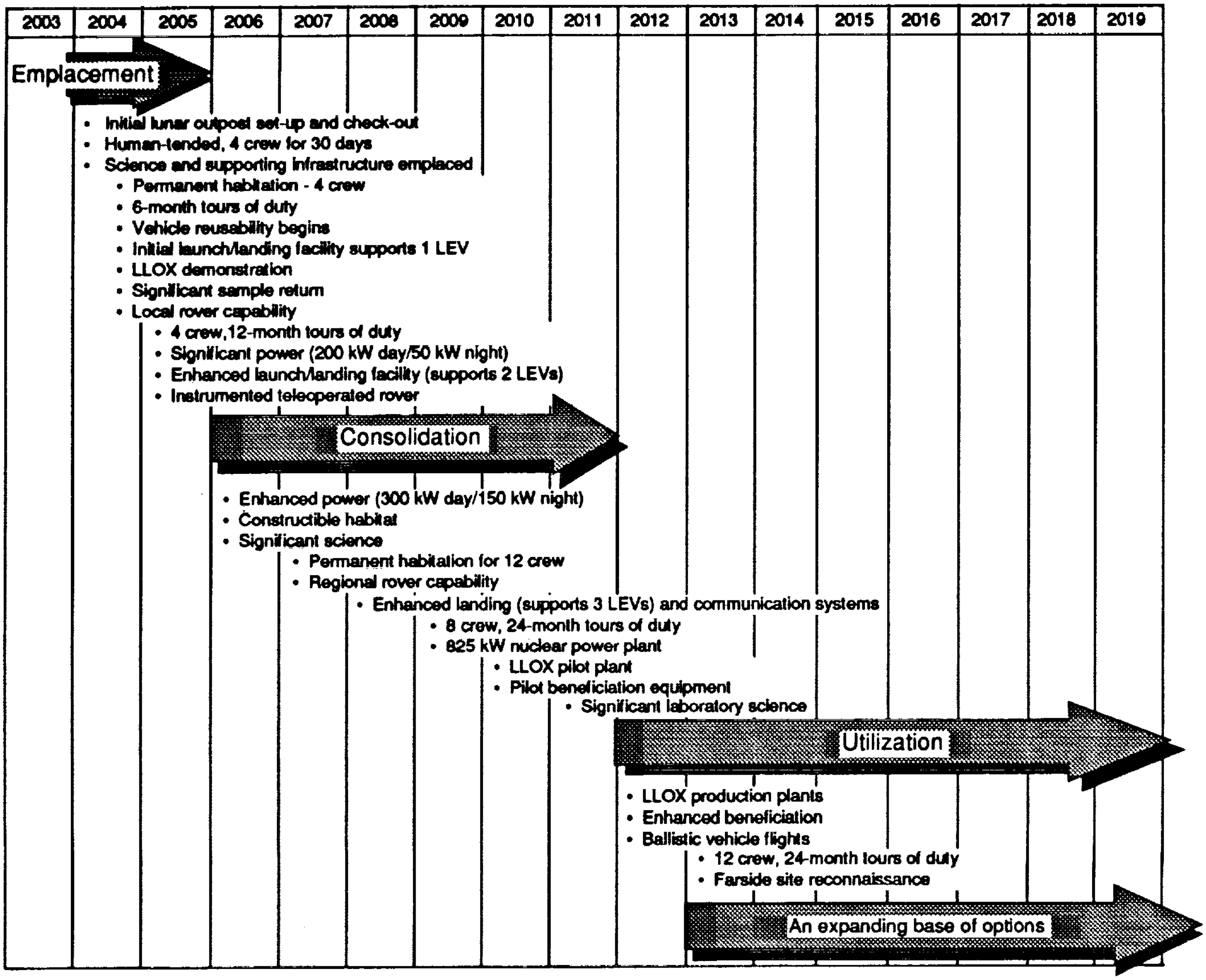


Figure 3.1.1-1.- Lunar outpost development phases.

portation infrastructure also is expanded during this phase, as confidence is established in both reusing the vehicles and servicing them at Space Station Freedom and on the lunar surface.

Utilization Phase. The objectives of this phase are to use in situ resources and to continue to expand experience in outpost habitation and operation. Emphasis shifts to making resources available for further exploration. Lunar oxygen production begins, and the oxygen is used for life support and to fuel the excursion vehicles based on the lunar surface. The territory of human operation now includes a human-tended facility on the lunar far-side and access to other points on the lunar surface. Further expansion of the outpost in this phase is limited, allowing other exploration missions to begin within a steady level of funding.

3.1.1.2 Mission Profile

The Lunar Observer spacecraft, currently planned to be launched in 1997, will support site selection and resource mapping of the lunar surface. For purposes of case study analysis, a site was chosen at 0 degrees latitude and 24 degrees east longitude, just north of the crater Moltke in the southernmost part of Mare Tranquillitatis. This site was selected because the study constrained the outpost to be located on the equator in order to ensure daily access to a low-lunar parking orbit. In addition, the hydrogen reduction of ilmenite technique was used as the lunar oxygen production process in the study, which limits potential sites to those in which high ilmenite basalts are found. Only two such areas exist on the lunar equator: a small one in Oceanus Procellarum and a larger one in southern Mare Tranquillitatis. The latter was

chosen because of its proximity to the Apollo 11 landing site.

Installation of the initial lunar habitation facilities begins in late 2003. From 2003 through 2005, three piloted missions and two unmanned cargo missions are flown to the lunar surface. The first crew tour of duty is 30 days, during which time the crew completes the deployment and begins operation of the initial support and habitation systems. A 6-month period of unmanned testing and verification of the surface facilities follows.

Permanent habitation of the outpost begins in mid-2004, when four crewmembers stay for 6 months, and the facilities are delivered to the surface to maintain the lunar excursion vehicle for personnel transfer. Habitation facilities are enhanced to provide more living and pressurized laboratory space. The number of permanent crew grows to eight in 2008. Crew tours of duty increase to 1 year in 2005 and to 2 years in 2008. This increase provides the opportunity to obtain valuable data in the areas of human adaptation to reduced gravity environments and long-term isolation from Earth, human performance degradation countermeasures technology, and Earth-independent outpost operational experience.

The outpost evolution continues as the number of permanent crew increases to 12 in 2012, and the outpost can support up to three lunar excursion vehicles (LEVs). Oxygen production, initiated in 2012, provides lunar liquid oxygen (LLOX) for the environmental control and life support systems as well as for propellant for LEVs. Science capabilities are expanded to include large pressurized laboratories for life sciences research and human biomedical studies, long-range pressurized rovers and ballistic excursion vehicles for human access to distant locations on the lunar surface, and large astronomical arrays.

The overall mission architecture is illustrated in figure 3.1.1-2. Earth-to-orbit transportation of mission cargo, vehicles, and propellant is accomplished by a combination of Shuttle-C and Block II Shuttle-C vehicles. Large payload shrouds are required for delivery of the reusable space transportation vehicles, whereas smaller shrouds are sufficient for cargo and propellant deliveries. In addition, delivery of the space transportation vehicles to low-Earth orbit occurs less frequently, as compared to cargo and propellant delivery, due to the assumed 10-year mission life of the reusable space transportation vehicles. A combination of Shuttle-C and Block II Shuttle-C vehicles, instead of a larger heavy lift launch vehicle, was chosen to provide the ETO transportation for the Lunar Evolution case study. The Shuttle-derived vehicles provide a suitable and efficient mix of vehicle, propellant, and cargo delivery while reducing the initial development costs and reaching a balance of payload size and on-orbit operations. The Shuttle-C payload capability to Space Station Freedom is assumed to be 71 t with a large-diameter payload shroud (10 m in diameter by 30 m in length) for lunar transfer vehicle/lunar excursion vehicle delivery. A smaller-diameter payload shroud (4.6 m by 15 m) is used for propellant and cargo delivery. The payload capability to Space Station Freedom for the smaller Block II Shuttle-C is assumed to be 61 t per flight. Six Shuttle-C launches per year are assumed to be available for the ETO delivery requirements. Support and mission crews are transported to Space Station Freedom and returned to Earth by the Shuttle.

Space Station Freedom is the staging location between Earth and the Moon for the lunar transfer vehicles (LTVs). Payloads and propellant are stored at Freedom, and Freedom concurrently provides the servicing facilities for the LTVs. In addition, Freedom houses lunar mission crews in transit, and provides housing for the LTV/LEV/payload processing support personnel.

Both cargo and piloted missions use LTVs for transfer from low-Earth orbit (LEO) to low-lunar orbit (LLO), insertion into LLO, and return to LEO. LEVs are used to transport cargo and crew to the lunar surface and from the lunar surface for rendezvous with the waiting LTV. Both vehicles are capable of being operated in an unmanned mode. The LTVs use an aerobrake on Earth return, arriving in an orbit from which the crew can be retrieved and transferred to Space Station Freedom. In addition, the translunar trajectory design permits a lunar flyby and free-return abort to LEO if necessary prior to lunar orbit insertion.

The mass that must be delivered to LEO, LLO, and the lunar surface in support of the Lunar Evolution case study is summarized in figures 3.1.1-3 through 3.1.1-5. The annual mass requirements to LEO fall within the capacity of six Shuttle-C and two Shuttle flights for all years except 2004, averaging 356 t per year for the first 10 years, and 180 t per year for steady-state operations. The mass requirements for 2004 can be slightly adjusted by delivering some cargo in 2003, thereby limiting the Shuttle-C flight rate to a maximum of six per year. This ETO flight rate, combined with 14 Space Shuttle flights per year (NASA's baseline launch rate), appears to be a natural break point above which major new facilities will be required at KSC. The number of crew and the duration of their stay are illustrated in figure 3.1.1-6, with each block of time representing one tour of duty. The adjusted flight rates needed to accommodate the mass and crew requirements to LEO and LLO are illustrated in figures 3.1.1-7 and 3.1.1-8. Major programmatic milestones are shown in figure 3.1.1-9.

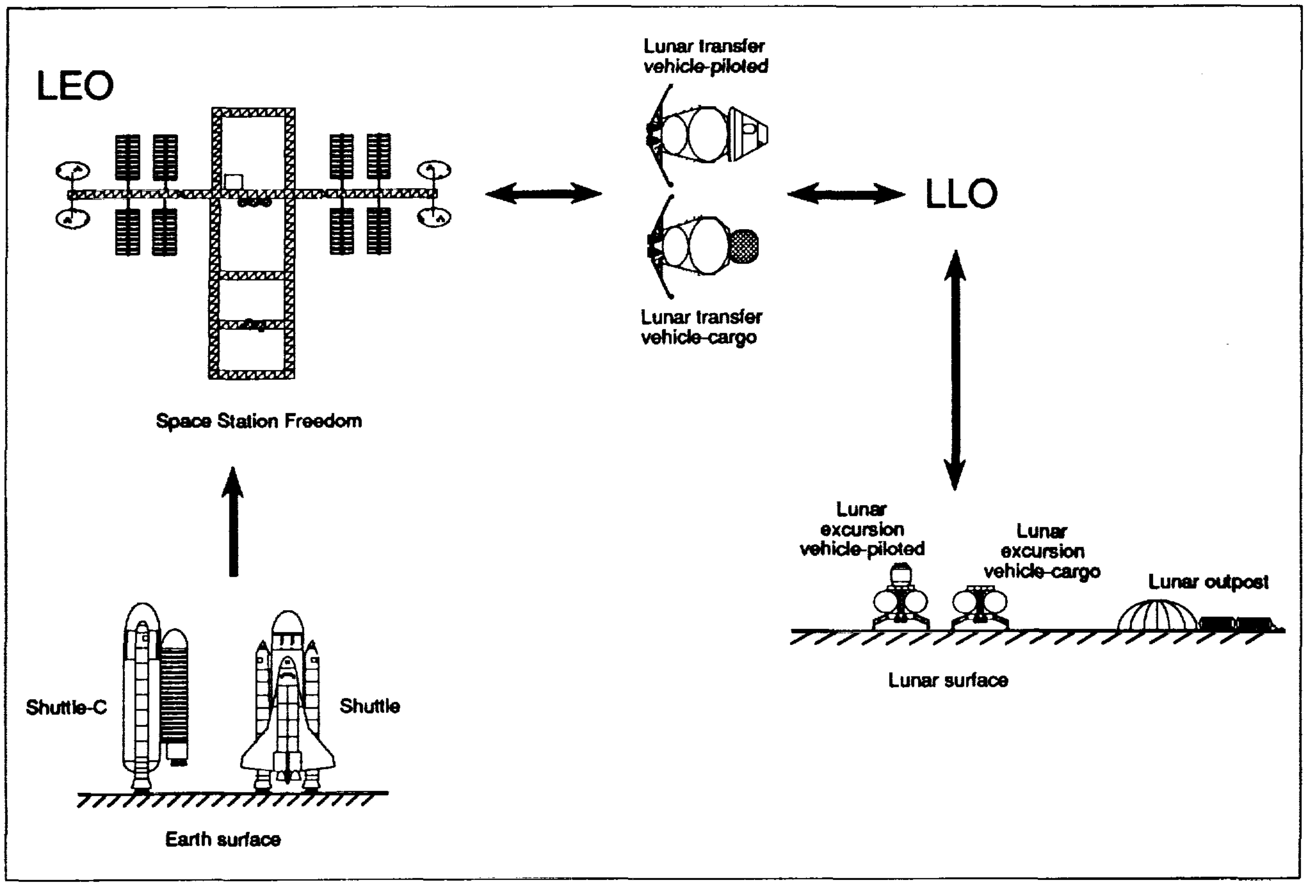
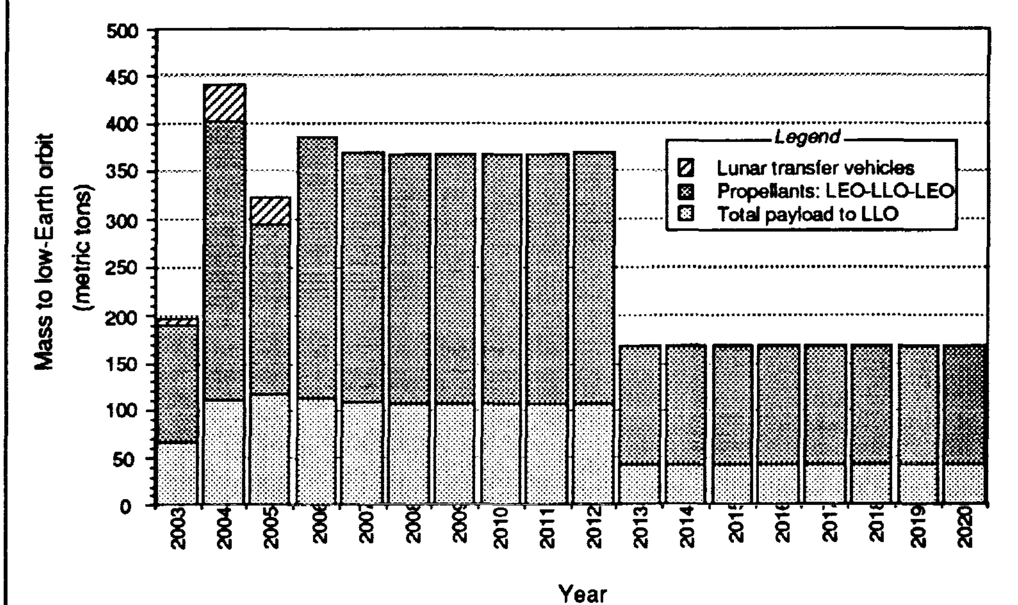


Figure 3.1.1-2.- Mission architecture for lunar outpost buildup.

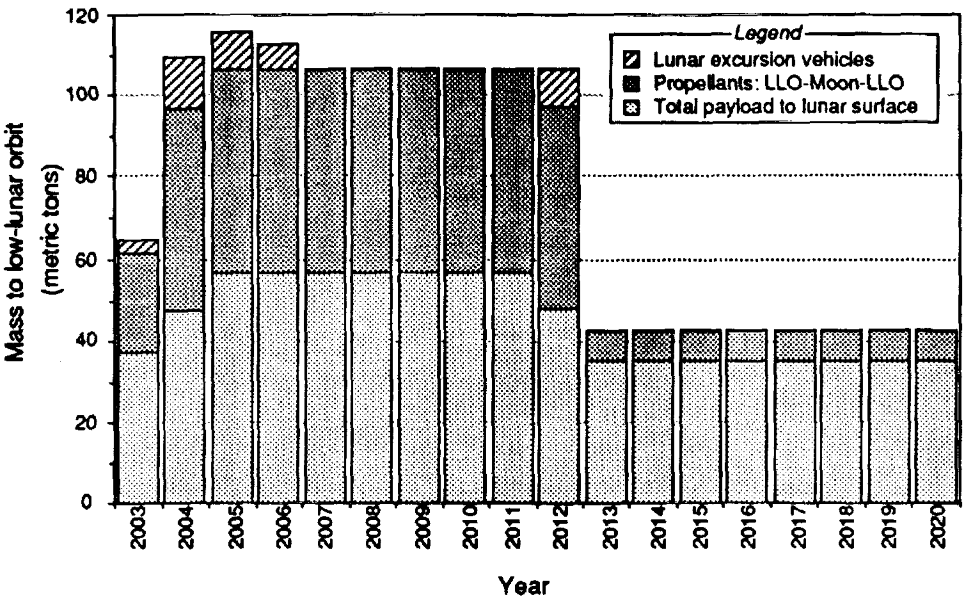


Lunar transfer vehicles include the dry LTV-Cs and LTV-Ps.

Propellants: LEO-LLO-LEO includes the propellants brought from Earth used by the LTVs in travelling between LEO and LLO.

Total payload to LLO includes crew and supplies, hardware, equipment, and LEVs and LEV propellants (and a payload margin to be delivered to the lunar surface).

Figure 3.1.1-3.- Lunar Evolution case study: mass to low-Earth orbit.

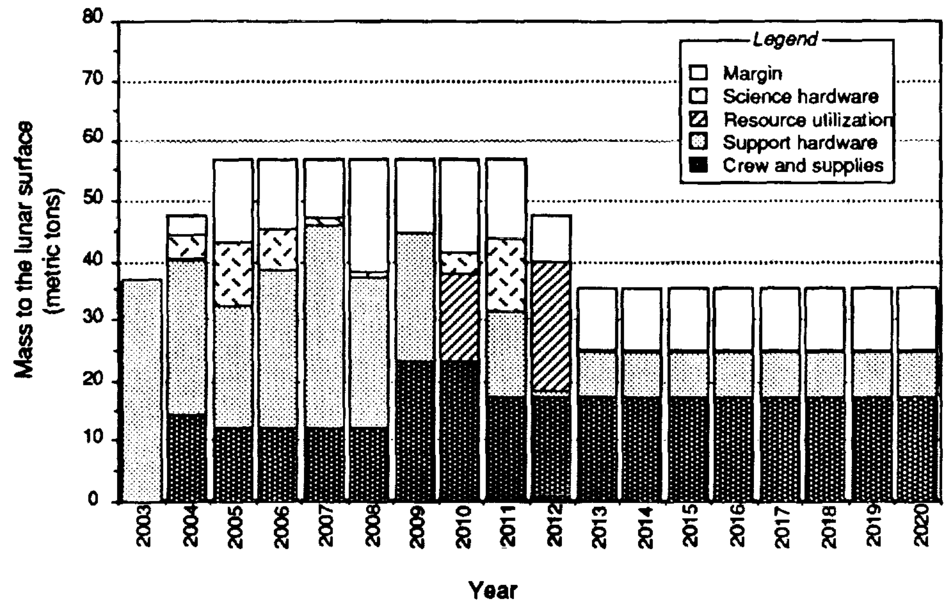


Lunar excursion vehicles include the dry LEV-Cs and LEV-Ps.

Propellants: LLO-Moon-LLO includes the propellants brought from Earth to be used in the LEVs in travelling between LLO and the lunar surface.

Total payload to the lunar surface includes crew and supplies, hardware, equipment, and a payload margin.

Figure 3.1.1-4.- Lunar Evolution case study: mass to low-lunar orbit.



Margin is the contingency mass to account for uncertainties in payload designs, tare, and unidentified spare parts.

Science hardware includes the equipment used to conduct the science experiments on the Moon (e.g., geophysical stations, geological exploration equipment).

Resource utilization includes the equipment used to obtain, transport, and process lunar resources.

Support hardware includes the equipment used to support the lunar outpost (e.g., habitation modules, rovers).

Crew and supplies includes the lunar outpost crew and all of their associated consumables and expendables.

Figure 3.1.1-5.- Lunar Evolution case study: total payload to the lunar surface.

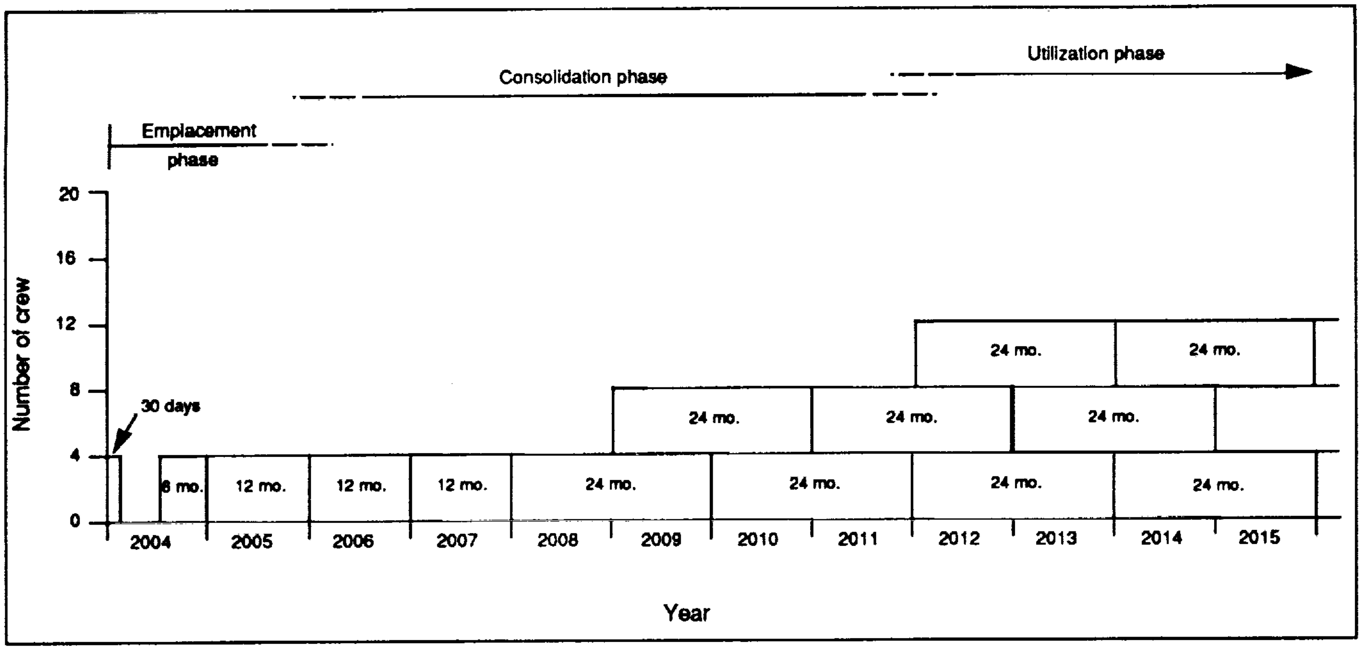
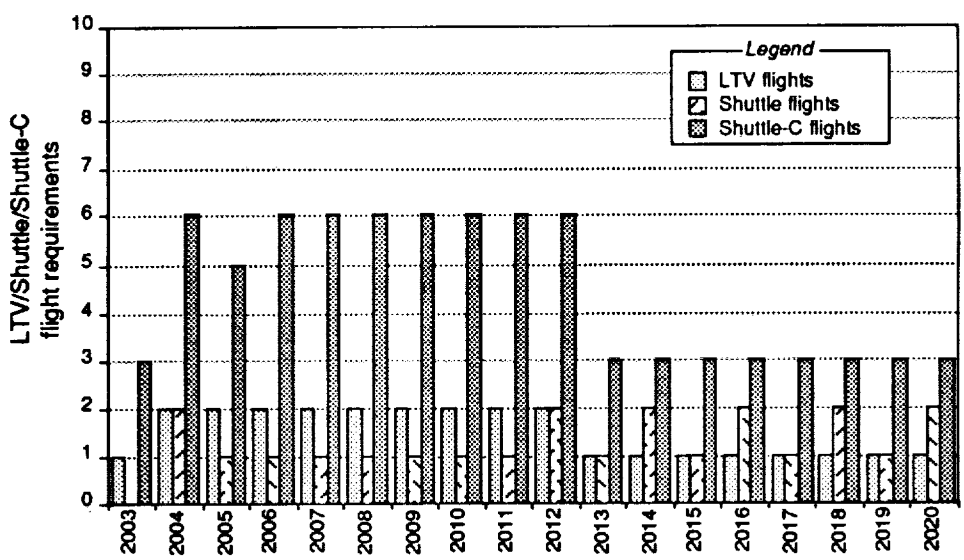


Figure 3.1.1-6.- Growth in number of crew and duration of stay at the lunar outpost.



LTV flights include the LTV-C and LTV-P flights from LEO to LLO delivering crew, payloads, and propellants for the lunar outpost.

Shuttle flights include the Shuttle flights from the Earth's surface to LEO delivering crew and payloads.

Shuttle-C flights include the Shuttle-C flights from the Earth's surface to LEO delivering payloads and propellants.

Figure 3.1.1-7.- Low-Earth orbit flight rates.

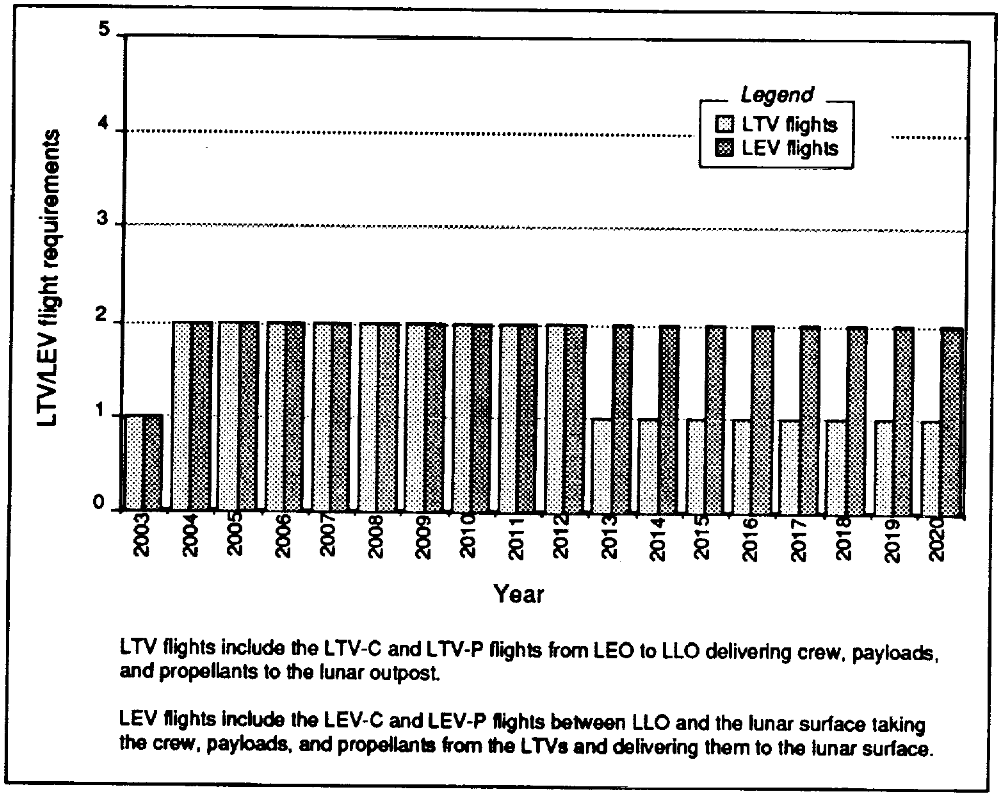


Figure 3.1.1-8.- Low-lunar orbit flight rates.

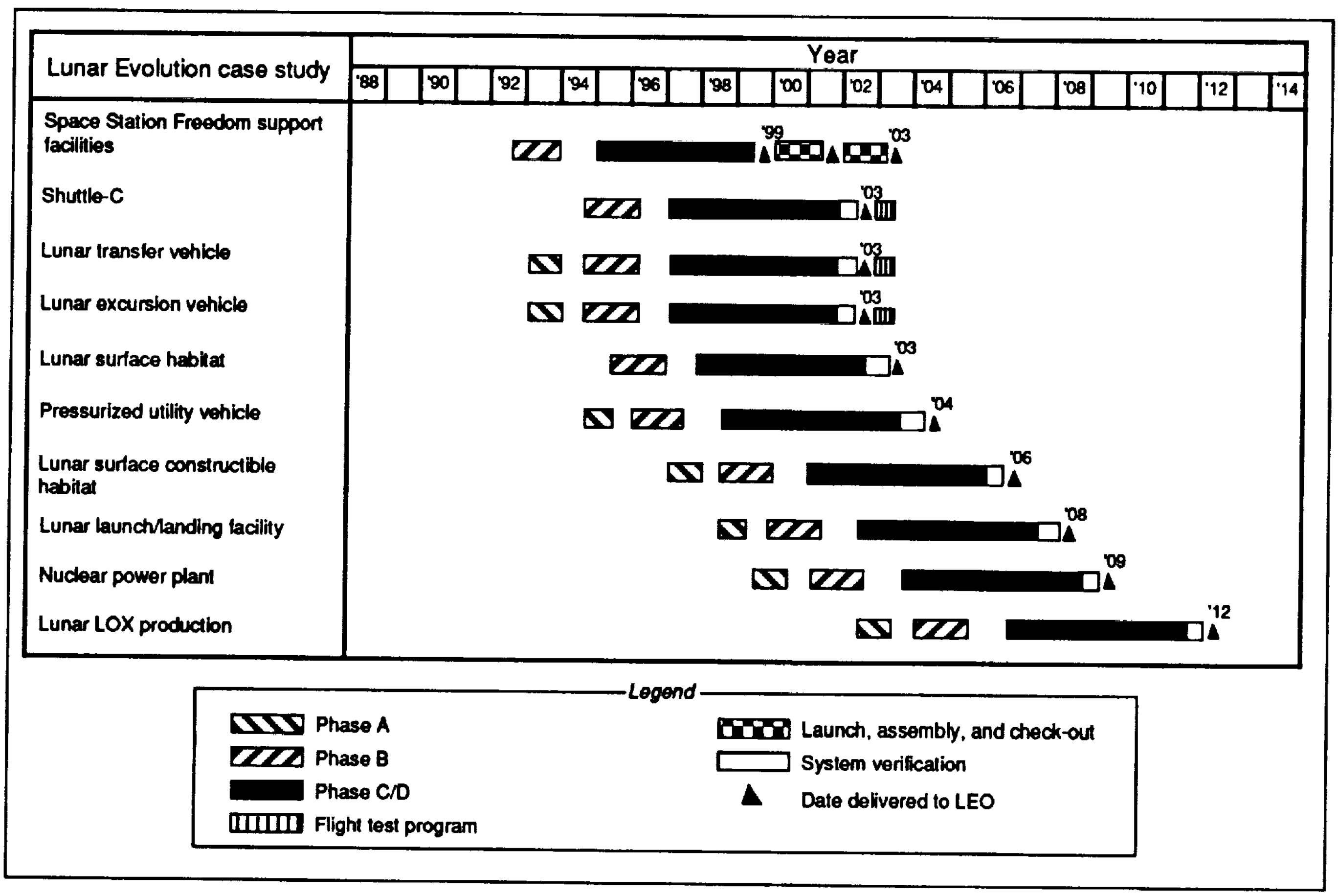


Figure 3.1.1-9.- Lunar outpost element development schedule.

3.1.2 Science Opportunities and Strategy

The science program described here has not received widespread scrutiny and approval by the science community, and is intended to reflect a preliminary approach to human lunar exploration within the specific framework and mission characteristics of the Lunar Evolution case study. The science program must still evolve through the normal maturation process of definition, refinement, review, and consensus within the science community. However, some benefit was derived from early efforts of various science strategy and oversight groups recently established and sponsored by OEXP. Versions of the MASE science program were made available for comments and input to OEXP, the Exploration Science Working Group, the Center Exploration Program Scientists, the Opportunity Analysis Group at JPL, and the newly formed Lunar Science Strategy Group at JSC.

The appropriate balance between manned and unmanned scientific exploration needs to be further examined. The science developed for this case study assumes a particular mix of the two, but future scenarios need to seriously consider what an optimal mix would actually be.

3.1.2.1 Opportunities

The lunar environment offers unique scientific opportunities, among which are the ability to study the Moon's current state, history, origin, and resource potential. Lunar geological and geophysical characteristics can be studied by conducting site surveys to collect rock, soil, and regolith samples to determine their compositions, distributions, and ages (dating analysis done in Earth-based laboratories) and to study the Moon's subsurface properties, morphology, stratigraphy, erosion phenomena (micrometeorites), dust mobility, cosmic rays, and regolith maturity.

Traverse science provides opportunities for mapping geologic formations on the regional scale, conducting geophysical surveys of lunar magnetic and gravitational fields, carrying out deep and shallow seismic experiments, and identifying resources and determining their distributions. Regional traverses provide the means to study impact cratering history, the age distribution of craters, and the relationship of lunar catastrophic events to mass extinctions on Earth. Regional surveys also calibrate and validate remotely sensed lunar orbital data.

Surface stations can be emplaced to monitor seismic activity and magnetic field variations, allow heat flow measurements to be made, and make solar wind and cosmic-ray measurements. Since the Moon passes through Earth's magnetic tail, particle and fields experiments can measure particle activity much more accurately than any Earth-orbital or Earth-based instruments. Surface monitoring stations can also observe the lunar "atmosphere" and ionosphere by studying the atmospheric composition and structure, its spatial and diurnal and long-term temporal variations, and possible atmospheric buildup attributed to human activity at a lunar outpost.

Using the Moon as an observing platform for astronomical and other outward-looking investigations has particular advantages. Lunar characteristics, including the absence of an atmosphere, slow rotation rate, and seismic stability, provide unparalleled conditions for optical and radio astronomy. The lunar environment opens new observing frequencies and allows the use of techniques such as very long baseline interferometry. In addition, the lunar farside is naturally insulated from radio signals from Earth, and, therefore, opens a new frontier for radio astronomy.

The Moon can be used as a platform for laboratories to conduct geochemical and petrological analyses of lunar materials in support of lunar science and of resource evaluation. Life sciences research in a lunar laboratory would include plant growth studies using hydroponics and lunar soil; study of animal and human performance, behavior, and physiology in the lunar environment; and observation of the biological effects of low gravity, radiation, and dust in a long-term lunar environment.

A physics laboratory on the Moon could study proton decay, relativity, lunar fluid mechanics, gravity, and the fifth force. In addition, such phenomena as Earth's magnetosphere can be studied not only on the lunar surface but also in transit to and from the Moon, a category of observations termed "cruise science." Transit times to and from the Moon are short, about 3 days, and cruise science would be limited to these magnetospheric studies.

A more detailed treatment of science opportunities is included in Volume VI of the FY 1989 Exploration Studies Technical Report.

3.1.2.2 Strategy

Lunar exploration begins locally near the landing site with traverses on foot and in unpressurized short-range rovers. Exploration extends to regional with the introduction of a pressurized rover and an instrumented unmanned rover. With the introduction of lunar ballistic vehicles in the utilization phase, global access to the lunar surface is provided, and exploratory expeditions are conducted to such sites as the lunar farside, poles, and other areas of scientific or resource interest.

Concurrent with the buildup of exploration range is the

progressive buildup of overall science capability. This buildup is moderated by mass limitations to the lunar surface from LEO, and is a function of the increasing availability of pressurized space that can be dedicated to laboratory use. Laboratory space is limited in the emplacement phase to the first habitation module and a dedicated science laboratory trailer. A geochemistry/petrology laboratory is initially emplaced in the trailer, providing an early analytical capability that can be used both for general rock and mineral analyses and for lunar resource evaluation.

During the consolidation phase, the second habitation facility is constructed with ample space to expand the geochemistry/petrology laboratory into a fairly comprehensive analytical capability. Laboratory space now provides for comprehensive life sciences research, introducing plant growth and microbial experiments as well as more sophisticated studies of animal behavior and human adaptations to the lunar environment.

Significant astronomical capability is gained, again growing gradually from a modest collection of optical and radio telescopes to a comprehensive collection of telescopes on the nearside and a low-frequency radio array deployed on the farside during the utilization phase, when ballistic expeditions are feasible.

A candidate mission for further exploration during the utilization phase would be a ballistic manned mission to the lunar polar areas for reconnaissance, geologic and geophysical exploration, and the search for the possible existence of water. A typical science payload for such a mission would include geologic exploration equipment, a rover equipped with an electromagnetic sounder, a geophysical station, and one or several monitoring telescopes.

Figures 3.1.2-1, 3.1.2-2, and 3.1.2-3 present a more detailed treatment of the specific science strategies and objectives, and list the science payloads for each of the

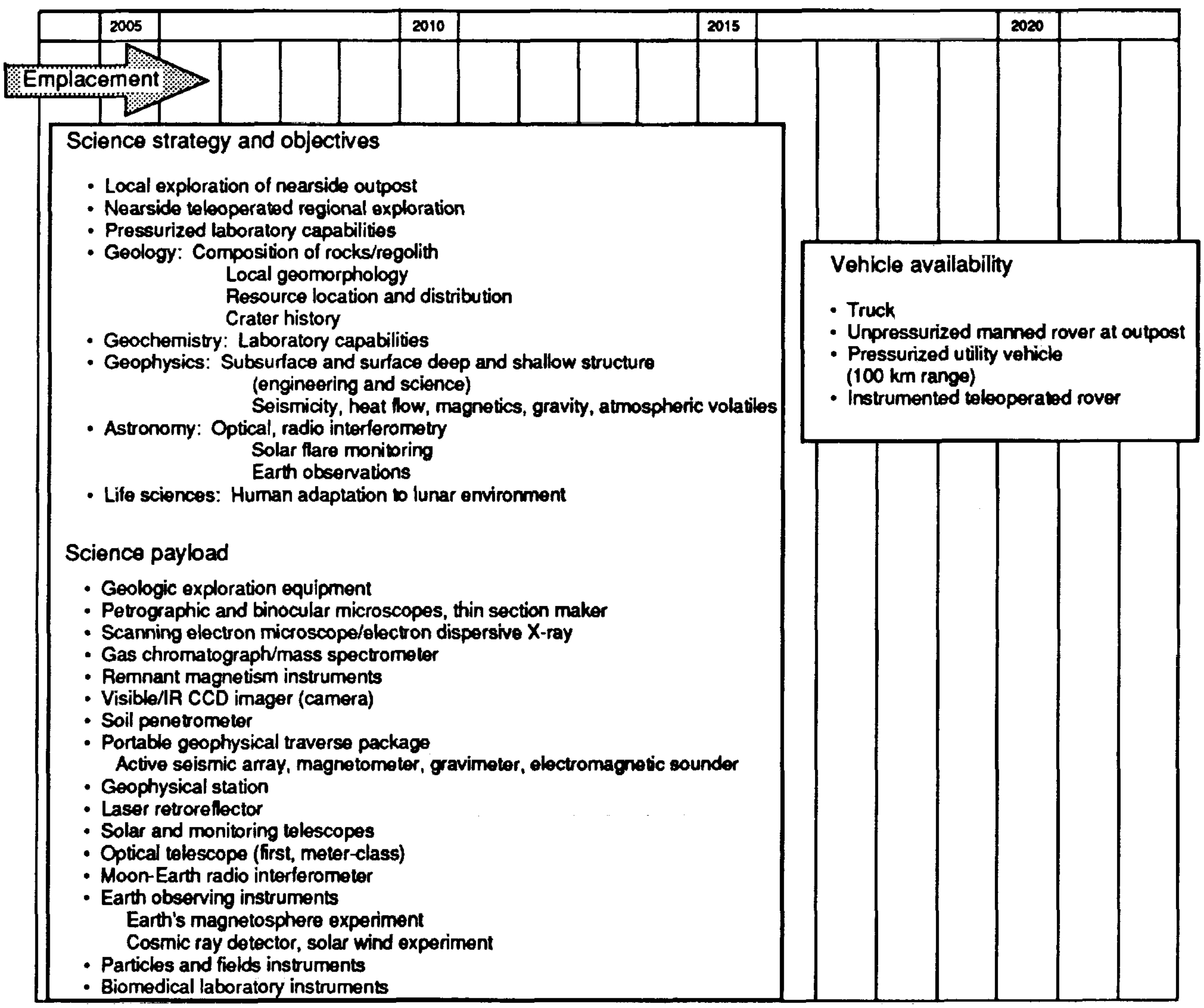


Figure 3.1.2-1.- Lunar science scenario: emplacement phase.

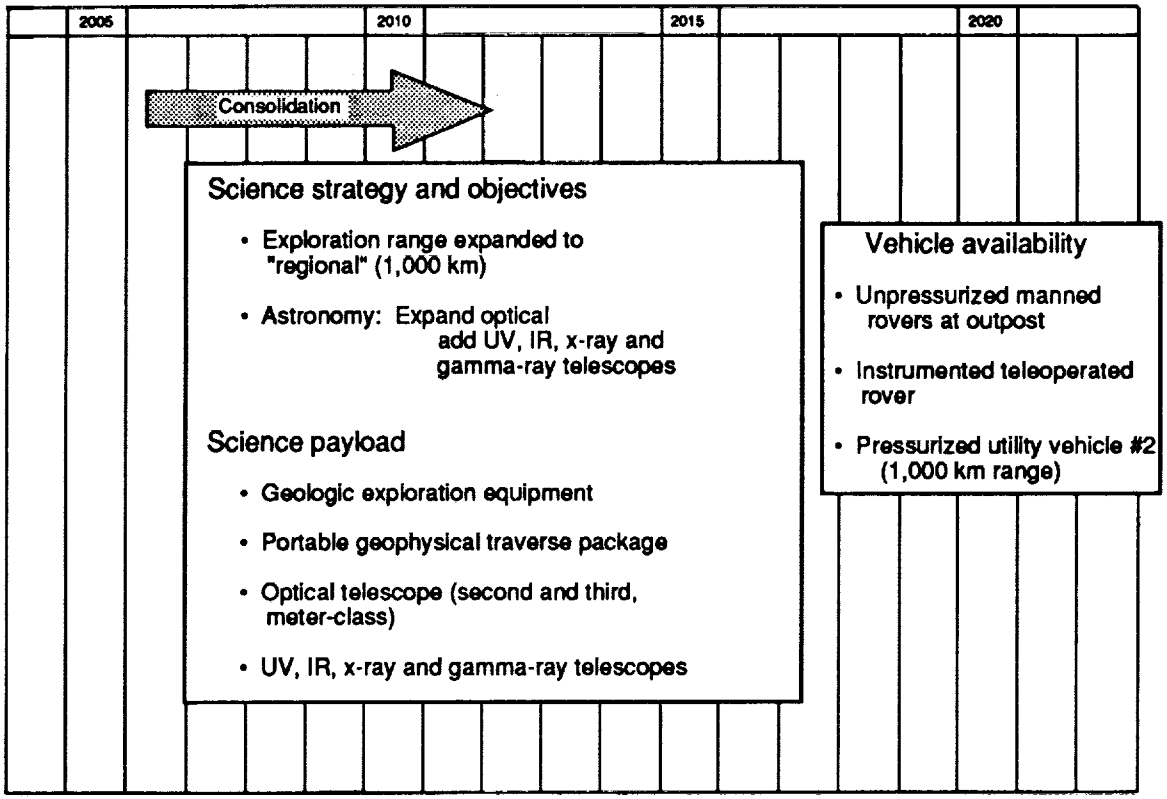


Figure 3.1.2-2.- Lunar science scenario: consolidation phase.

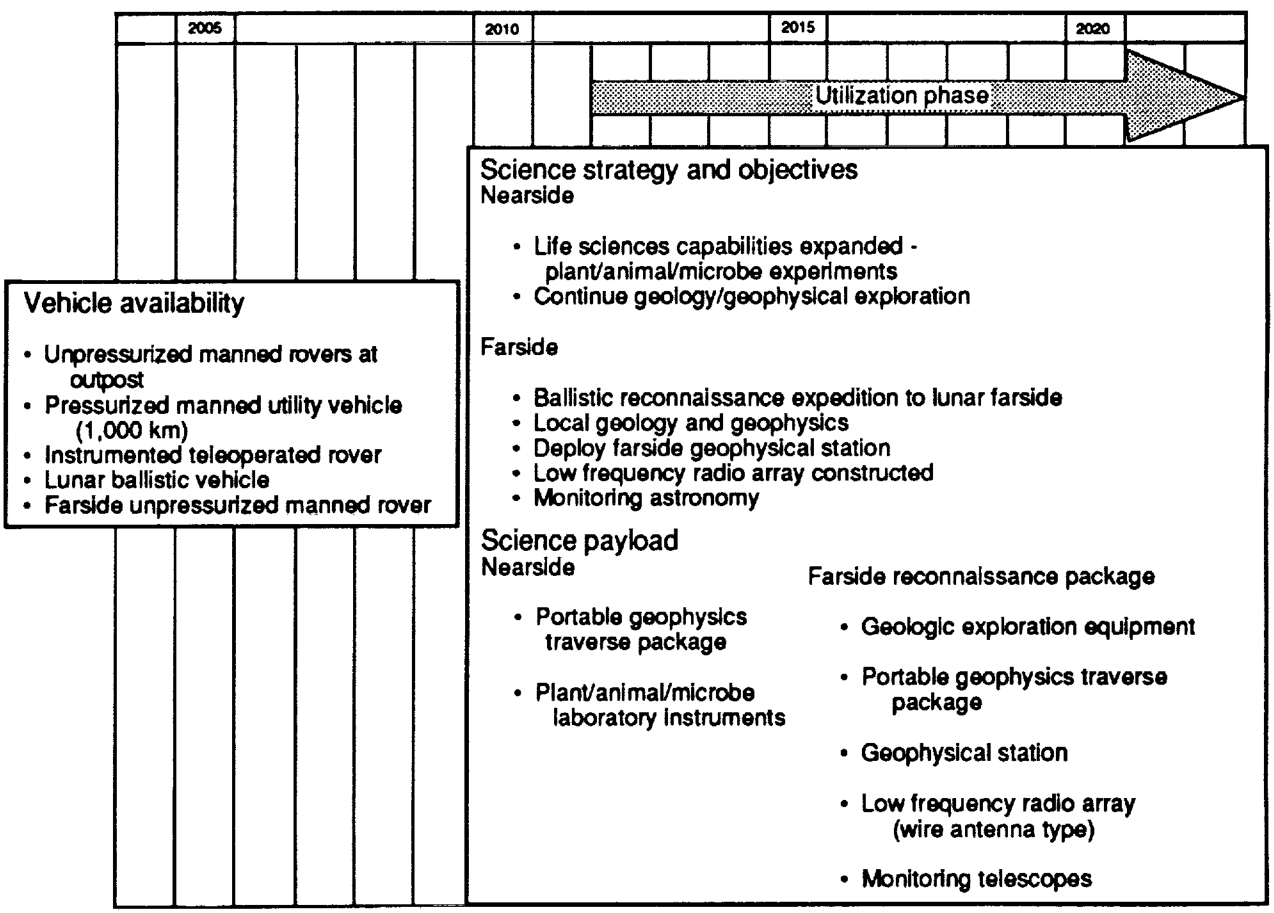


Figure 3.1.2-3.- Lunar science scenario: utilization phase.

three phases of the case study. Each phase has specific science objectives, a strategy to accomplish the objectives, and an associated payload to carry out the observations.

The science payload for the Lunar Evolution case study is accommodated by phasing the delivery of the instruments to the lunar surface over several flights within each of the outpost phases.

3.1.3 Transportation Systems

The transportation system for the Lunar Evolution case study consists of two classes of vehicles: (1) the lunar transfer vehicle (LTV), used to transfer personnel and/or cargo between LEO and LLO, and (2) the lunar excursion vehicle (LEV), used to transfer personnel and/or cargo between LLO and the lunar surface. Personnel vehicles, which can accommodate up to eight crewmembers, are designated LTV-P or LEV-P, whereas cargo vehicles are designated LTV-C or LEV-C. All vehicles are designed to be reusable with the LTVs based and serviced at Space Station Freedom and the LEVs based and serviced on the lunar surface at the outpost. The vehicles have an assumed reuse lifetime of 10 missions.

3.1.3.1 Elements and Systems

An important design concept of this case study was the extensive use of vehicle commonality. Common LEVs that can be used for either a pure cargo mission or a combined personnel and cargo mission keep the lunar outpost overhead to a minimum. A single vehicle design requires a smaller inventory of spare parts, fewer systems for the crew to understand, and the ability to interchange vehicle parts. Also, a common design eliminates the need for separate backup vehicles for personnel or cargo missions, thus minimizing the number of LEVs on the lunar surface and providing inherent redundancy and mission safety. Commonality of LTVs has the same effects at Space Station Freedom.

The LTV uses liquid oxygen and liquid hydrogen propellants. A large foldable aerobrake on the LTV allows the returning vehicle to use Earth's atmosphere for orbit capture in LEO, thus reducing the total propellant required for a lunar mission. Since the aerobrake is foldable, it requires no on-orbit assembly at Freedom. When an LTV is used for a cargo mission, only unmanned payloads are loaded onto the LTV and delivered to lunar orbit for transfer to the LEV and subsequent delivery to the lunar surface. When an LTV is used for a personnel mission, a 9 t crew module is attached to the LTV, and both crew and payloads are delivered to LLO. The module is an ablative, Apollo-style design capable of Earth entry and a soft landing. This design provides a redundant capability for Earth return of the crew in the event of an aerobrake failure.

The LEV uses the same engines and propellants as the LTV. The engines on the LEV employ a smaller nozzle than their LTV counterparts, which results in a lower engine specific impulse. However, the smaller nozzle allows the LEV to have short landing legs and reduced dry mass. When an LEV is used for a cargo mission, only unmanned payloads are loaded onto the LEV and delivered to the lunar surface. For a personnel mission, a crew module is attached to the LEV, and both crew and payloads are delivered to the lunar surface. These LTV and LEV configurations are shown in figure 3.1.3-1.

The payload delivered and propellant used by each vehicle depends on the operational mode of the vehicle. Table 3.1.3-I summarizes the payload and propellant loadings for the LTVs and LEVs during each mission phase, and the following paragraphs describe their operational use.

Although the LTVs and LEVs are designed to be reusable and space-based, there is currently no experience in servicing vehicles in space. Therefore, the initial three LTVs and three LEVs used for the emplacement phase are not planned for reuse; instead, they are expended and used as test-beds to better understand what types of servicing will be required and how this servicing will be done. As operational and maintenance experience is gained, vehicles can gradually be reused to greater degrees until they become totally reusable.

During these initial flights, a typical cargo mission consists of an LTV-C departing LEO with a fully loaded and fueled LEV-C as payload. Once in LLO, the LEV-C separates from the LTV-C and descends to the lunar surface. When the cargo vehicles are operated in an expendable mode, the LTV-C is left in LLO for a subsequent controlled deorbit to the lunar surface; the LEV-C is left on the lunar surface. Since the LTV-C does not return to LEO, an aerobrake and return fuel are not required, thereby increasing the payload capacity of the vehicle. Similarly, since the LEV-C only descends to the lunar surface, no ascent propellant is required, and the mass of this return propellant is substituted for by payload.

The personnel missions during the emplacement phase (see figure 3.1.3-2) utilize an LTV-P departing LEO with a fully loaded and fueled LEV-P as payload. Once in LLO, the LEV-P separates from the LTV-P and descends to the lunar surface. After the crew have completed their mission on the lunar surface, the LEV-P lifts off from the lunar surface to rendezvous with the LTV-P in LLO. When the personnel vehicles are operated in an expendable mode, the LEV-P descends to the lunar surface after the crew has transferred to the LTV-P, and the LTV-P is expended in LEO after returning the crew to Free-

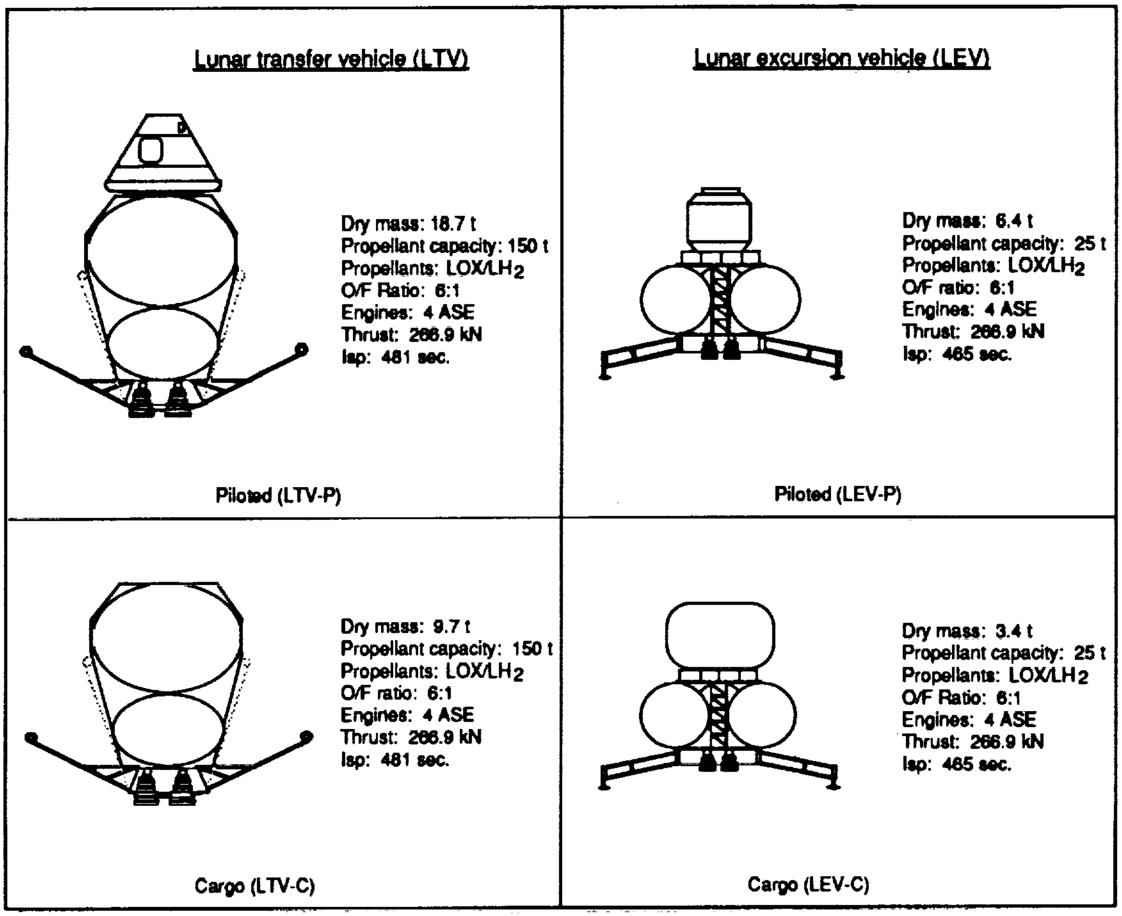


Figure 3.1.3-1.- Lunar Evolution vehicle configurations.

TABLE 3.1.3-I.- LUNAR EVOLUTION VEHICLE LOADINGS

Mission	Vehicles used	Operational mode		Vehicle dry mass (t)		Propellants required (source: amount (t))		Payload carried (t)	
		LTV	LEV	LTV	LEV	LTV	LEV	LTV	LEV
Initial cargo mission (no aerobrake on LTV)	1 LTV-C 1 LEV-C	Expended in LLO	Expended on lunar surface	7.8	3.4	Earth: 124.1	Earth: 24.2	64.6	37.0
Consolidation phase cargo misson	1 LTV-C 1 LEV-C	LEO based	Lunar surface based	9.7	3.4	Earth: 125.8	Earth: 24.9	57.9	33.0
Utilization phase cargo mission using LLOX in LEVs	1 LTV-C 2 LEV-Cs	LEO based	Lunar surface based	9.7	Per LEV-C: 3.4	Earth: 106.8	Per LEV-C: Moon: 20.4 (LLOX) Earth: 3.4 (LH_2)	46.8	Per LEV-C: 20.0
Initial personnel mission	1 LTV-P 1 LEV-P	Expended in LEO	Expended in LLO	18.7	6.4	Earth: 146.6	Earth: 24.8	54.7	23.5
Consolidation phase personnel misson	1 LTV-P 1 LEV-P	LEO based	Lunar surface based	18.7	6.4	Earth: 135.7	Earth: 24.8	48.5	23.7
Utilization phase personnel mission using LLOX in LEVs	1 LTV-P 1 LEV-P 1 LEV-C	LEO based	Lunar surface based	18.7	LEV-P: 6.4 LEV-C 3.4	Earth: 124.8	LEV-P: Moon: 21.2 (LLOX) Earth: 3.5 (LH_2) LEV-C: Moon: 20.4 (LLOX) Earth: 3.4 (LH_2)	42.1	LEV-P: 15.2 LEV-C: 20.0

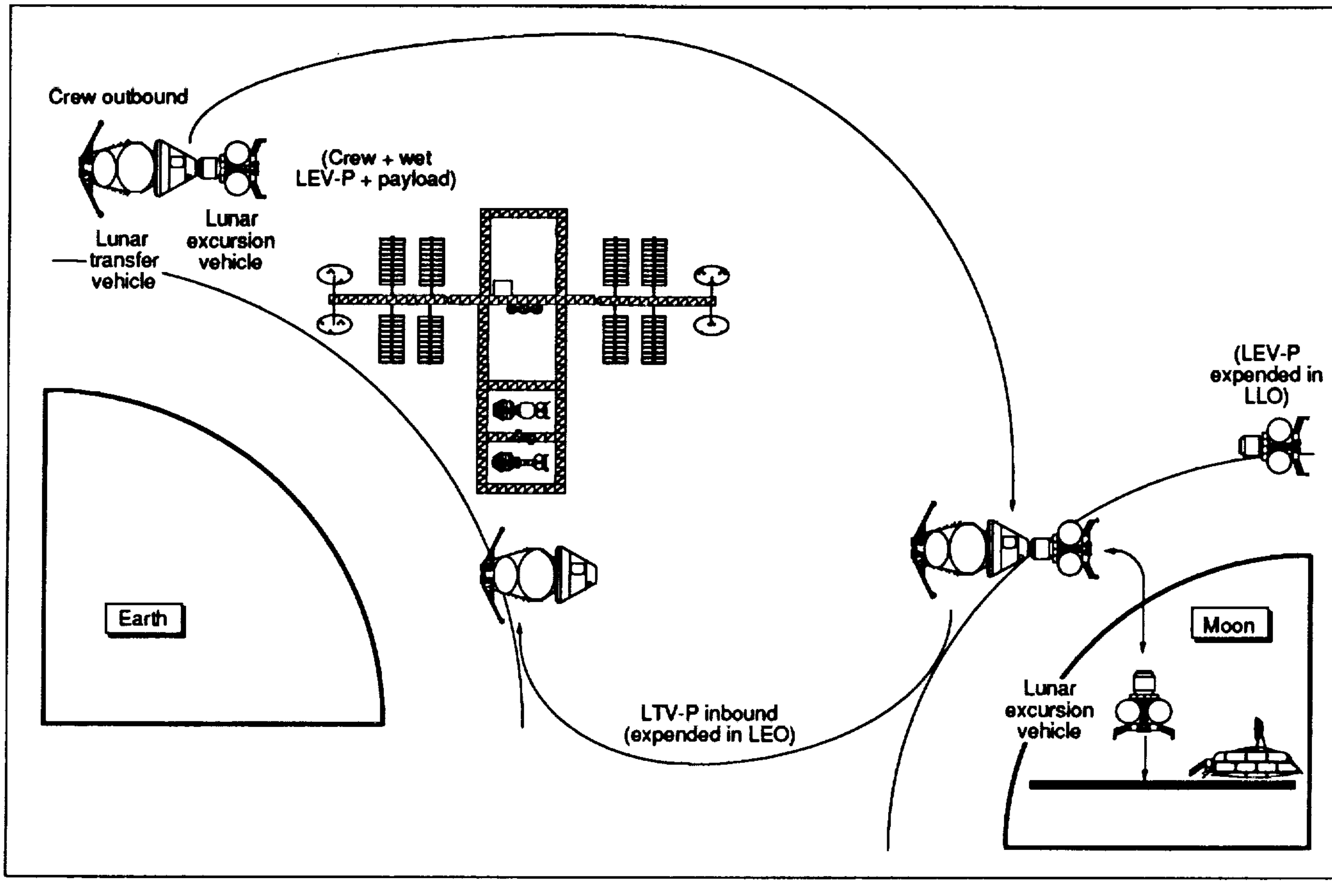


Figure 3.1.3-2.- Personnel mission during emplacement phase.

dom. On the first personnel mission, the LTV-P remains in LLO for the 30 days the crew is on the lunar surface. On the second personnel mission, once the LEV-P separates from the LTV-P and descends to the lunar surface, the LTV-P returns unmanned to LEO where it is expended by functioning as a reusability test-bed. After the 6-month crew stay, the LEV-P returns the crew to LLO where they rendezvous with a third personnel mission arriving from LEO. The second LEV-P is expended in LLO, while the third LEV-P (the first consolidation phase mission) descends to the lunar surface with a new crew. The third LTV-P returns the (last emplacement phase) 6-month lunar outpost crew to LEO. A cargo mission during this phase is similar except that the LEV-C is expended on the lunar surface and the LTV-C is expended in LLO.

As experience is gained in servicing vehicles in space and confidence is achieved in using vehicles for multiple missions, an operational transition occurs in which the expended missions are replaced by fully reusable missions. These missions are characterized by basing and servicing the LTVs at Space Station Freedom and the LEVs on the lunar surface. To conduct these missions, the necessary launch/landing and servicing facilities must be in place and operational at Freedom and the lunar outpost.

During the consolidation phase, reusable vehicles are used for both cargo and personnel missions. For a cargo mission, an LTV-C launched from LEO will rendezvous in LLO with an LEV-C launched from the lunar surface. After rendezvous, payloads and propellant are transferred from the LTV-C to the LEV-C. The LEV-C then descends to the lunar surface, and the LTV-C returns to Space Station Freedom. The vehicles are then serviced and prepared for the next mission.

A personnel mission during the consolidation phase is illustrated in figure 3.1.3-3. Personnel missions consist of an LTV-P, which is launched from LEO to rendezvous in LLO with an LEV-P launched from the lunar surface. After rendezvous, crew, payloads, and propellant are transferred. The LTV-P delivers crew and payloads for the LEV-P and the propellant required by the LEV-P to take the crew and payloads to the lunar surface and return the crew to LLO.

As the outpost continues to evolve and expand, lunar resource utilization becomes an integral part of its capabilities. Part of this expansion includes producing oxygen on the lunar surface from indigenous lunar resources. This lunar liquid oxygen (LLOX) is then used as the oxidizer for the LEVs as well as the oxygen supply for the life support system. Since the LEVs are using

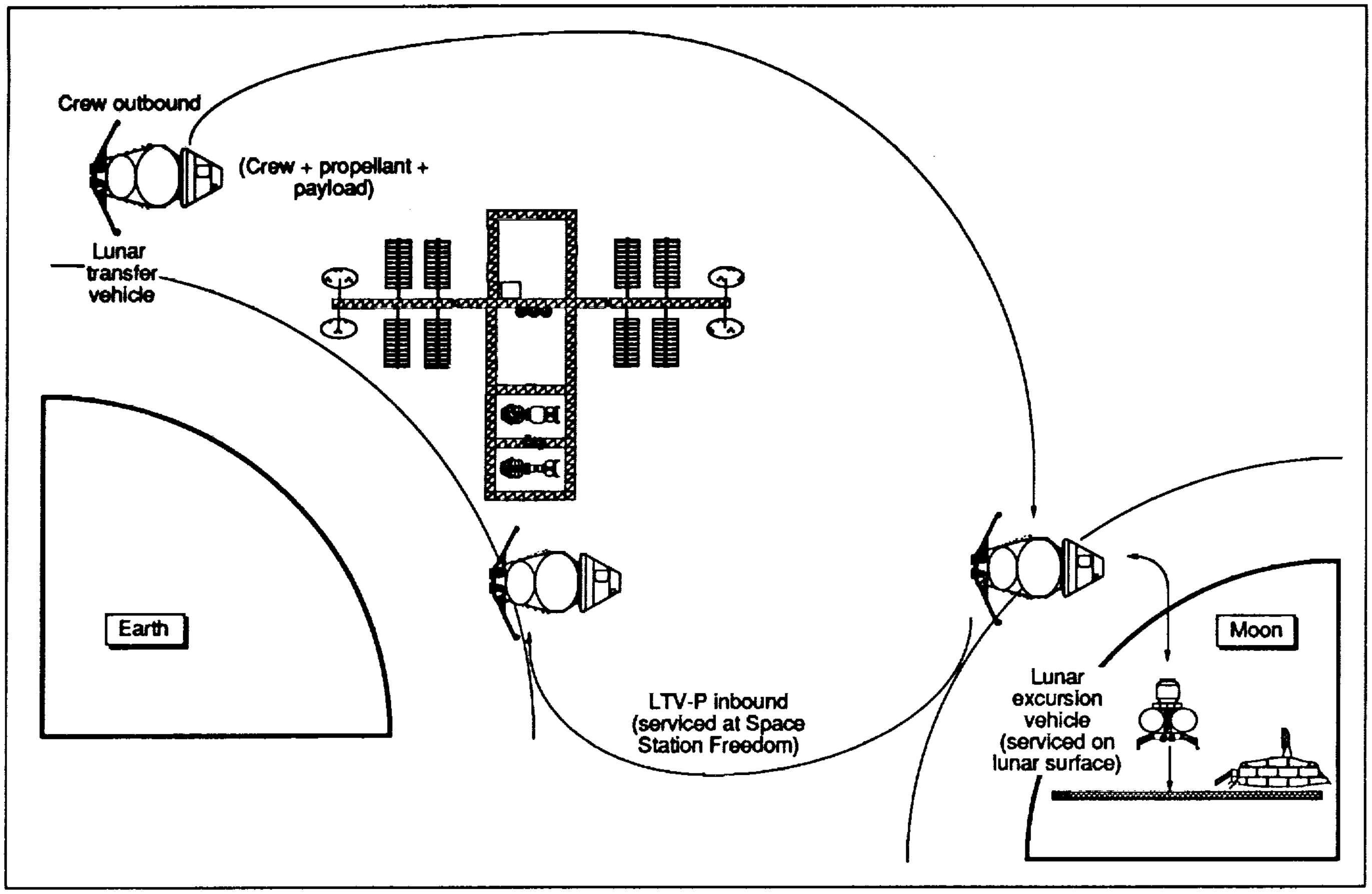


Figure 3.1.3-3.- Personnel mission during consolidation phase.

LLOX, the LTVs can use their propellant capacity to deliver two LEV payloads from LEO, effectively cutting the LTV flight rate in half. Since no hydrogen is being produced on the lunar surface, the LTVs must still deliver Earth-supplied hydrogen required by the LEVs.

A mission in which LLOX is an integral part (figure 3.1.3-4) consists of an LTV-P launched from LEO to rendezvous in LLO with both an LEV-P and an LEV-C launched from the lunar surface. After this rendezvous, transfer of crew, payloads, and Earth-supplied hydrogen takes place. In addition to crew and payload, the LTV-P provides enough hydrogen to the LEV-P to deliver the crew and payload to the lunar surface and return the crew to LLO. The LTV-P also provides payload and the hydrogen required by the LEV-C to deliver its payload to the lunar surface and return to LLO empty. A cargo mission during this phase would be similar to the LTV-P mission; i.e., an LTV-C would rendezvous in LLO with two LEV-Cs.

3.1.3.2 Enabling Technology

The vehicle designs used in this case study rely on key technologies that must be developed, tested, and proven acceptable. The LTV design depends on currently unproven aerobraking technology for capture into LEO. The aerobrake is a lightweight, flexible/ceramic design consisting of a fixed ceramic tile central disk and an outer annulus of deployable ceramic cloth supported by ribs. In addition, all vehicle concepts use advanced chemical propulsion systems with Advanced Space Engines and low boiloff cryogenic propellant tanks. Also, vehicles are assumed to be based and serviced at Space Station Freedom and on the lunar surface. (For this case study, the term "serviced" refers only to maintaining a vehicle's state of health, which includes keeping avionics at proper temperatures, minimizing propellant boiloff, performing minor repairs, and protecting the vehicles from the local environment). Vehicles are assumed to be reliable enough to be used for 10 missions before replacement becomes necessary.

Vehicle rendezvous, docking, and transfer operations require several advances in technology. For cargo missions, the rendezvous, docking, and transfer operations in LLO occur without humans in the local vicinity, which implies either autonomous or teleoperated maneuvers. For personnel missions, the rendezvous, docking, and transfer operations in LLO occur without any crew EVA. For all missions, the transfer of cryogenic propellants occurs in LEO, LLO, and on the lunar surface.

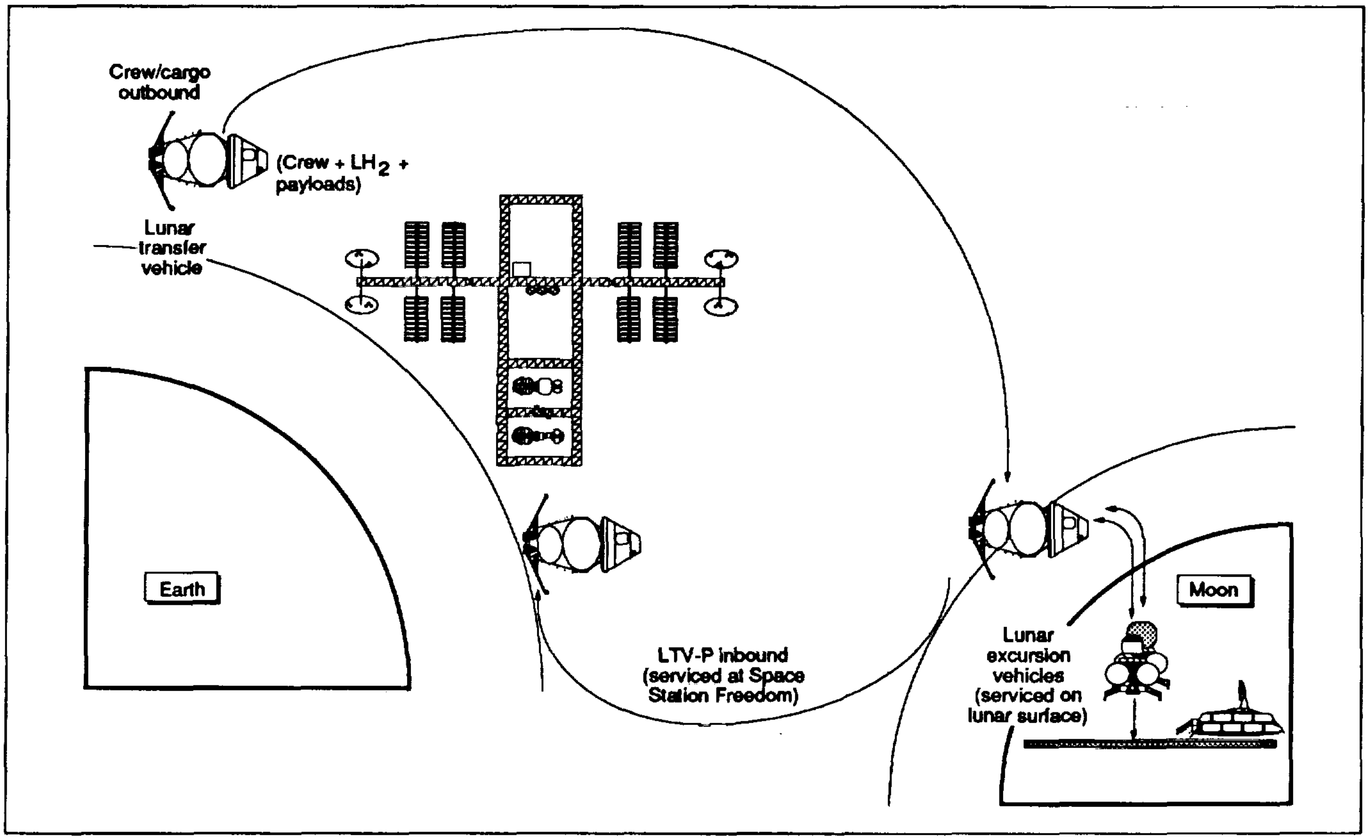


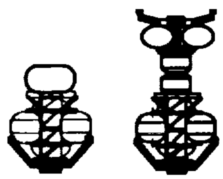
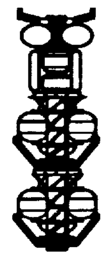
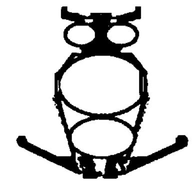
Figure 3.1.3-4.- Personnel mission with LLOX being used in lunar excursion vehicles.

3.1.3.3 System Alternatives

An important trade conducted during this case study addressed the sizing of the LTVs. The demands on the Earth-Moon transportation system evolve as the outpost itself evolves from the emplacement phase through the utilization phase. Early in the emplacement phase, all LTVs and LEVs are expended, which places the burden on the LTV to transport an LEV, along with its propellant and cargo, to LLO for each mission. Conversely, during the utilization phase, when LLOX is available for fueling the reusable LEVs, the LTV only delivers the LEV's cargo and hydrogen to LLO for each mission. Thus, the LTV's payload to LLO during the utilization phase is only half that required for the emplacement phase flights or for any mission in which an LEV needs to be delivered to LLO. The LTV design alternatives studied were sizing the LTVs for the utilization phase, when LEVs are using LLOX, versus sizing the LTV for the initial flights of the emplacement phase. Figure 3.1.3-5 illustrates three sizing approaches and the issues associated with each.

For the caravan and staging sizing approaches, the LTVs were designed to deliver to LLO one LEV payload and enough hydrogen for one LEV mission. Although this approach results in one LTV and one LEV flight per mission for the utilization phase, the limited payload mass capabilities of the LTVs dictate that two LTVs be used to deliver a fully loaded and fueled LEV during the initial flights of the emplacement phase. Also, during the remainder of the emplacement phase and throughout the consolidation phase, two LTVs are still required to deliver one LEV payload and enough LEV propellant (oxygen and hydrogen) for one LEV mission. Thus, the majority of the missions during this case study would have a 2:1 LTV to LEV flight ratio per mission. As shown in figure 3.1.3-5, these missions can be accomplished using either a caravan or staging option. In the caravan mode, one LTV would deliver the cargo and LEV hydrogen to LLO, where it would rendezvous with a second LTV, which had carried out the LEV fueled with oxygen. In this mode, the two LTV flights would depart LEO separately. The staging option uses two LTVs, serially staged, to deliver the LEV with its fuel and cargo to LLO. The basic problem with the small LTV in either the caravan or staging option is that it places the most difficult operational demands on the Earth-Moon transportation infrastructure at the beginning of the outpost's development, when little or no experience exists. Complex autonomous rendezvous, docking, and payload transfer operations must occur on the first lunar mission with the caravan option, whereas simultaneous processing of two LTVs per lunar mission is necessary for both caravan and staging options. Thus, the small LTV sizing option greatly impacts the activity level and

Vehicle stack at trans-lunar injection

Caravan	Staging	Scaled-up
		

- Issues during initial flights

Caravan	Staging	Scaled-up
Complicated initial operations 2:1 LTV / LEV ratio Vehicle processing doubled Early autonomous rendezvous Early autonomous payload transfer Small diameter aerobrake	Simplified initial operations 2:1 LTV / LEV ratio Vehicle processing doubled No early autonomous rendezvous No early autonomous payload transfer Small diameter aerobrake	Simplest initial operations 1:1 LTV / LEV ratio Simplified vehicle processing No early autonomous rendezvous No early autonomous payload transfer Larger diameter aerobrake

- Issues during utilization phase

Caravan	Staging	Scaled-up
Simplified utilization phase 1:1 LTV / LEV ratio	Simplified utilization phase 1:1 LTV / LEV ratio	Optional utilization phase 1:1 or 1:2 LTV / LEV ratio

Figure 3.1.3-5.- Lunar vehicle sizing options.

size of the servicing facility at Space Station Freedom at the very beginning of the case study, since two LTVs need to be concurrently processed for each mission to LLO.

To decrease the number of LTV flights, and thus decrease the associated LTV activity at Freedom, a scaled-up LTV was considered that would always allow for a 1:1 LTV to LEV ratio per mission. The larger LTV would also allow for a gradual increase in capability at Freedom, where the initial missions to the Moon could be accommodated with a less complex single LTV facility. The scaled-up LTV was sized to deliver a fully loaded and fueled LEV from LEO to LLO. Once LEVs are based on the lunar surface, one LTV delivers one LEV payload and enough LEV propellant (oxygen and hydrogen) for one LEV mission.

During the later portions of the utilization phase when the LEVs are using LLOX, the larger LTV has the ability to deliver two LEV payloads and enough hydrogen for two LEV missions, or it can continue to deliver enough payload and hydrogen for one LEV. Thus, the LTV to LEV ratio could be 1:2 or 1:1 for this portion of the case study. Because of the less complicated LEO operations associated with a 1:1 LTV to LEV ratio and the fact that fewer larger LTVs are needed for the case study than smaller LTVs, the scaled-up LTV design was selected.

3.1.4 <u>Orbital Node Systems</u>

Space Station Freedom serves as the staging location between Earth and the Moon for this case study. Operations conducted at Freedom include payload and propellant transfer, vehicle checkout and verification, vehicle refurbishment and storage, and life sciences research. Some modifications to Space Station Freedom will be required to meet the objectives of the case study.

3.1.4.1 Elements and Systems

Each LTV/LEV combination can be brought to orbit on a single Block II Shuttle-C launch, which minimizes or eliminates the need for on-orbit assembly. However, payload and propellant transfer will still be required. The need for a propellant storage facility is highly dependent on the flight rate to the Moon and available ETO launch capability. No long-term propellant storage is required for the Lunar Evolution case study, and no propellant storage facility is needed for the given flight rates.

The Space Station Freedom configuration for accommodating the lunar missions is shown in figure 3.1.4-1. Launch of the Freedom growth elements can be accomplished in 10 flights of a standard Shuttle-C vehicle. The configuration of Freedom for this case study includes enhanced truss and power capability (225 kW total power), habitation for the support crew, vehicle, customer, and intravehicular activity servicing facilities. Micrometeoroid and debris shielding is provided on the vehicle accommodation facility to protect the lunar vehicles and EVA support crew. This facility is designed to permanently accommodate two LTVs (simultaneously), with each vehicle requiring up to 3 months for complete processing between flights. The size and design of Space Station Freedom growth are highly dependent on the lunar flight rate and vehicle design.

With a Shuttle-C launch frequency of six per year, the processing, mission operations, and recovery can be accomplished in less than 90 days. Total traffic through Space Station Freedom averages two LTV flights to the Moon per year. This low flight rate will still allow vital research and development operations at Space Station Freedom to continue uninterrupted for up to 180 days.

3.1.4.2 Enabling Technology

Since reusable vehicles are assumed in this case study, the primary technological need is based upon the level of vehicle processing and refurbishment at Space Station Freedom. The degree to which the aerobrakes for the LTVs must be refurbished is currently unknown, but this operation could require technology advances. If the flight rate to the Moon is sufficiently high, a wet/hazardous cryogenic propellant storage facility may be required. The processing, storage, and handling of these propellants at Freedom are of extreme importance. The ability to process hazardous vehicles on-orbit will require advances in automation and telerobotics.

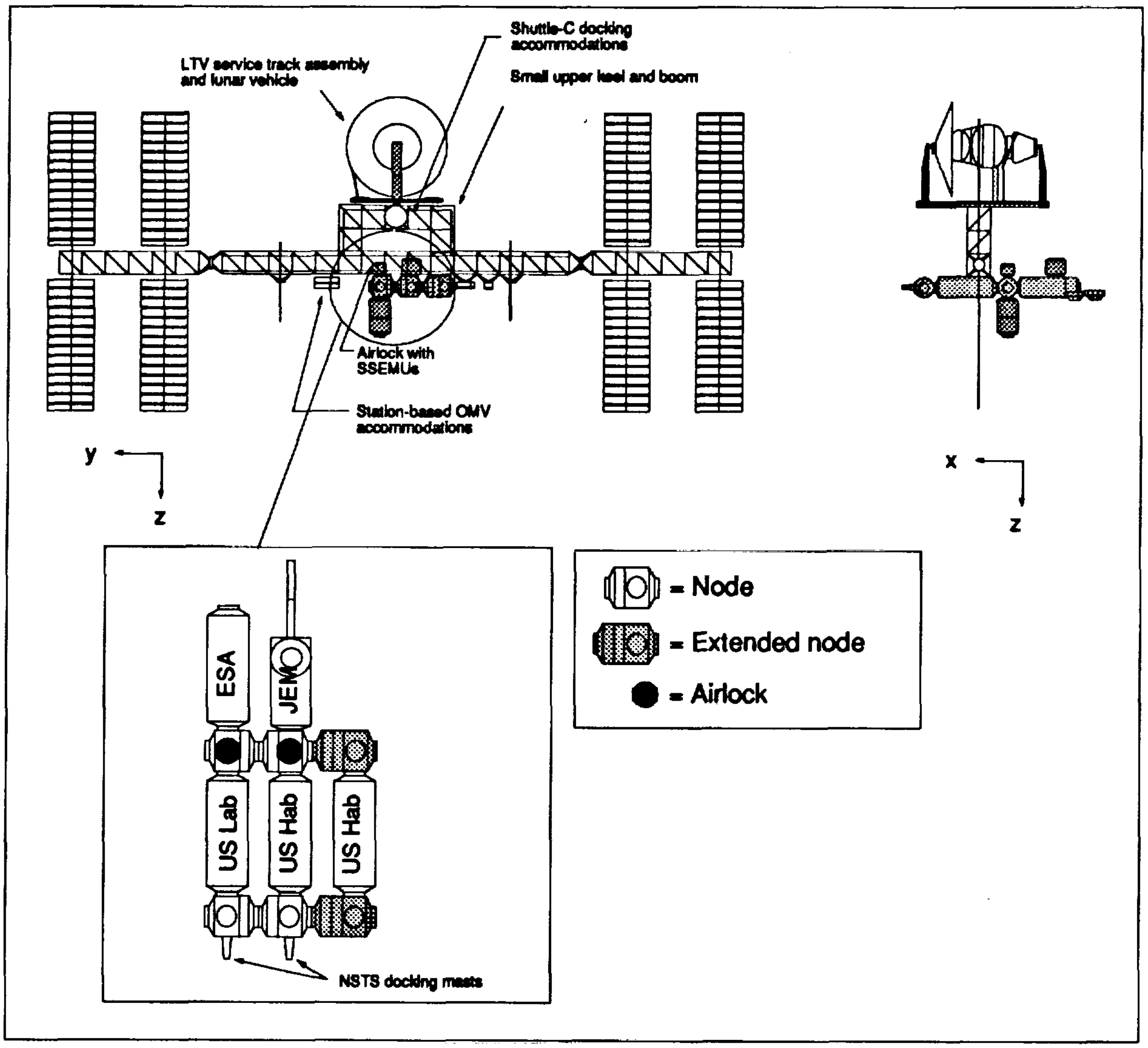


Figure 3.1.4-1.- Lunar vehicle accommodation at Space Station Freedom for LTV verification.

3.1.4.3 System Alternatives

Several growth options and alternatives for Space Station Freedom are currently being considered. These options include providing a simple vehicle staging "porch" (see section 4.2.3) for initial flights to the Moon in which vehicles are not reused; therefore, vehicle processing is not required. The only operations that must be conducted include vehicle checkout, crew ingress, and orbital maneuvering vehicle (OMV) transportation to the LEO propellant transfer point. This simple porch design provides the required functions for the early lunar missions while minimizing the impacts on Space Station Freedom, but it must be enhanced to provide adequate facilities for vehicle processing/refurbishment for the later LTV flights.

Another option provides for propellant transfer at Freedom while maintaining distance between the crew habitation and propellant storage facilities. In this design option, (discussed in section 4.2.3.4), tethers induce acceleration at the propellant depot, allowing for propellant settling in the tanks.

3.1.5 Planetary Surface Systems

The surface systems designed and the surface strategies employed for the Lunar Evolution case study provide for an outpost that gradually evolves from a small, human-tended systems test-bed and science station to a larger, permanently occupied outpost with facilities to support significant scientific exploration and research and the use of lunar resources. As the outpost develops, its capabilities for self-support increase, and the outpost begins to operate independently in areas such as lunar excursion vehicle servicing, lunar resource utilization, and day-to-day operations. This move toward self-sufficiency relieves some of the burden on Earth-based and LEO-based infrastructure and makes resources available so that human exploration missions to other solar system destinations can be planned and carried out.

The lunar outpost develops through three phases: emplacement, consolidation, and utilization. The emplacement phase is used to establish human presence on the Moon with a minimum of crew and infrastructure, and depends significantly on Earth support for crew supplies, construction teleoperation, and operations monitoring. During this phase, the outpost is capable of sustaining a crew of four for 6 months to 1 year, and lunar science is limited to the immediate vicinity of the outpost.

The habitation architecture and outpost capabilities are substantially enhanced in the consolidation phase. The larger lunar outpost enables the crew to perform at an increased level of activity with less Earth dependence. During this phase, the outpost is capable of sustaining a crew of eight for 1 year tours of duty, and lunar science is expanded to include pressurized laboratory investigations and the exploration of areas tens to hundreds of kilometers from the outpost.

The utilization phase is the last phase of development of the lunar outpost in this case study. The phase is instrumental in proving the usefulness of local resources and the viability of outpost self-sufficiency. During this phase, the outpost is capable of sustaining a crew of 12 for 2-year tours of duty, and emphasis is placed on a curtailed, yet sustained level of lunar operations. The combination of outpost self-sufficiency and curtailed lunar activities results in less support required from Earth and LEO, thus allowing the next phase of exploration to begin. Figure 3.1.5-1 shows the layout of the completed lunar outpost.

3.1.5.1 Elements and Systems

Habitation. As is the case with all the elements described in this section, the habitation facility was designed to allow for a gradual expansion of capabilities. The initial portion of the habitation facility consists of a Space Station Freedom-derived module modified for one-sixth gravity. The module is pre-outfitted at Earth and has everything necessary, except crew consumables, to sustain a four-person crew on the Moon for 30 days to 1 year. The initial habitat is delivered to the Moon on an unmanned cargo vehicle and emplaced on the lunar surface telerobotically from Earth. When the first crew arrives at the Moon, they need only connect the power supply and certify the module for pressurized occupation. The initial outpost is spartan at best, and very little space is available for science users due to the volume limitations. The telerobotic emplacement and capability for quick human occupation are two important characteristics of the initial outpost.

As the outpost develops, requirements for pressurized volume for crew accommodations, laboratory equipment, work space, and storage increase substantially. The habitation facility is expanded to accommodate these requirements by the addition of a large, spherical constructible habitat. The habitat takes about 1 year to construct (within the constraints of LEV flight frequency), and requires tasks such as excavating the site, erecting and pressurizing the structure, installing life support and thermal control systems, outfitting the interior, and covering the outside with regolith for radiation protection. The constructible habitat has several levels and can accommodate 12 people. The constructible also has enough habitable volume to provide several laboratories and a lunar outpost command center, giving the lunar crew the ability to make scientific and operational decisions at the outpost. This is an impor-

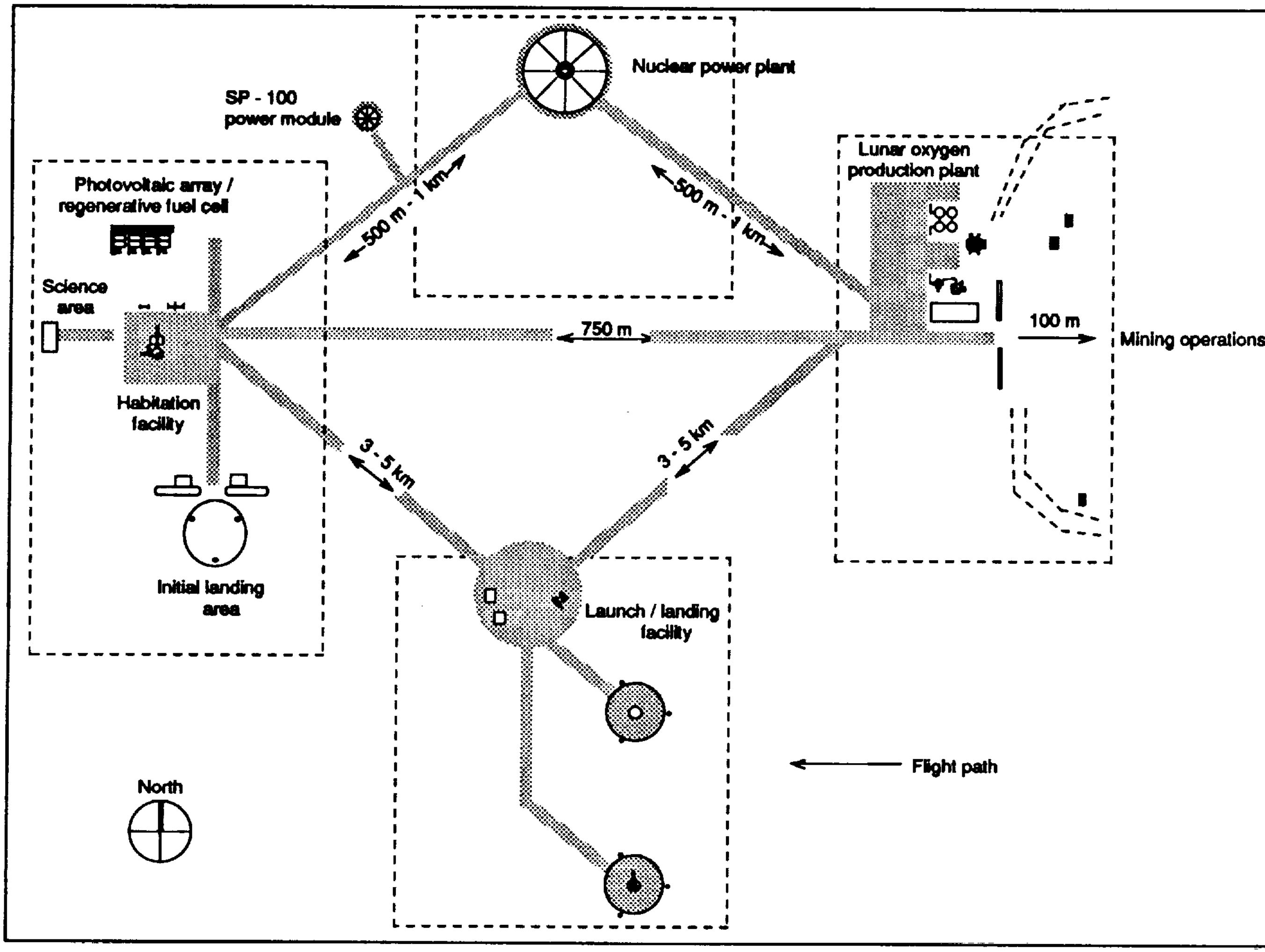


Figure 3.1.5-1.- Completed lunar outpost.

tant first step toward self-sufficiency. Figure 3.1.5-2 shows the completed habitation facility at the lunar outpost.

Life Support/Thermal Control Systems. The life support system (LSS) and thermal control system (TCS) are the two major systems that keep the habitation facility safe for the lunar outpost crew. The LSS selected for the lunar outpost is a regenerable physical-chemical system derived from the Space Station Freedom LSS. The initial LSS is integrated into the module and recovers roughly 96 percent of the system's water and oxygen. The advanced LSS in the large constructible habitat improves on the initial LSS by recovering materials from solid wastes, having the ability to use lunar-derived oxygen, and reducing the resupply of system expendables.

The thermal loads on the habitation facility require the TCS of the lunar outpost to be an active system. The TCS selected is similar to the Space Station Freedom TCS: two single-phase water loops for the acquisition of metabolic and equipment heat loads. The waste heat is rejected to the lunar environment by vertically mounted radiators oriented parallel to the plane of the solar ecliptic.

Radiation Protection. Radiation protection is a major concern for long-term habitation of extraterrestrial surfaces. The primary hazards are from solar flares and lengthy exposure to galactic cosmic radiation (GCR). Solar flares occur sporadically and are roughly correlated with the 11-year sunspot cycle. GCR contains more energetic particles than solar flares but at substantially lower fluxes. Solar flares can be lethal over short time periods, whereas GCR presents a long-term hazard. Recent studies indicate that for crew stay times of less than 1 year, GCR protection is not required. Thus, due to the short stay times of the lunar crews during the emplacement phase, the initial Space Station Freedom-derived module is not provided with GCR protection. However, solar flare protection is provided by a heavily shielded area at one end of the module. Lunar regolith can be used as shielding material to protect the habitation facility from radiation, and as outpost capabilities permit,

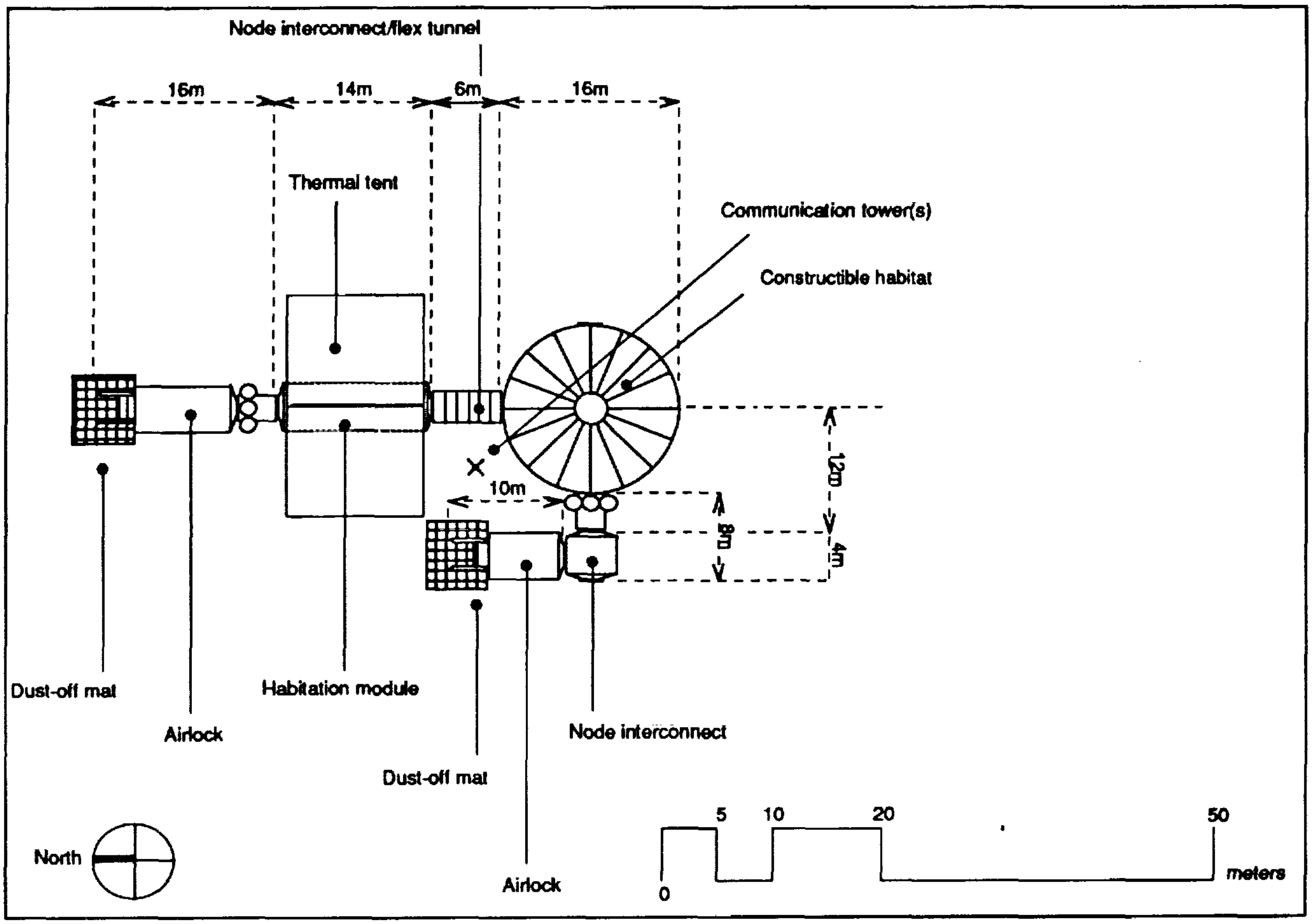


Figure 3.1.5-2.- Lunar outpost habitation facility.

the module is eventually covered with regolith and treated as a safe haven. As crew stay times increase, GCR protection becomes necessary, and the constructible habitat is covered with a half-meter of regolith during its construction. (This is a major difference from the results presented in the OEXP's FY 1988 Exploration Studies Technical Report, in which several meters of lunar regolith were required over the entire habitation facility regardless of crew stay times.)

Power. The expanding nature of the lunar outpost dictates an evolutionary approach to the power generation system. The initial power demands of the outpost are small and are accommodated by a photovoltaic array/ regenerative fuel cell (PVA/RFC) system. The PVA provides all the power for lunar daytime activities and also charges the RFC. The RFC provides the outpost with power during the lunar night. The initial PVA/ RFC system is telerobotically deployed near the outpost, so when the first lunar crew arrives they only have to connect the power system to the habitat.

As the habitation facility grows, the power demand increases due to the larger habitable volume, pressurized science laboratories, and the addition of lunar excursion vehicle servicing. The power demand is above the level where PVA/RFC systems are desirable; therefore an SP-100 nuclear power source with an improved specific power (power produced/system mass) is emplaced on the lunar surface. The SP-100 supplies a continuous power level of 100 kW throughout the lunar day and night, but has to be placed away from the outpost to protect the crew from radiation.

The continuous power level of the SP-100 satisfies all the outpost's power needs until lunar resource utilization begins, which requires power in the megawatt range. To supply this power, an 825 kW power source is added to the outpost's power generation system. This large nuclear reactor is based on SP-100 technology and uses dynamic power conversion. It is also located away from the outpost to protect the crew from radiation.

Launch/Landing Facilities. A major requirement associated with the Lunar Evolution case study is to maintain and service the LEVs at the lunar outpost. Accordingly, launch and landing systems are manifested to provide LEV turnaround on the lunar surface. LEV support equipment includes items such as thermal control carts to dissipate heat generated by the LEVs, a reliquefaction facility to capture propellant boiloff, light-weight thermal blankets to shield the LEVs, and a propellant

refill vehicle to transfer propellant between the LEVs and outpost storage facilities. The launch and landing of the LEVs are also supported by pad support equipment such as navigational beacons and aids, a range safety system, an electrical grounding system, and blast protection devices. Figure 3.1.5-3 shows the lunar outpost's launch and landing facilities.

Lunar Oxygen Production Plant. The Lunar Evolution case study examines the potentially substantial savings and enhancements to be obtained through the utilization of lunar resources. The high costs of delivering materials to the Moon from Earth drives the desire to use local resources. Lunar-derived liquid oxygen (LLOX) can be produced on the Moon by processing the lunar regolith, which is roughly 44 percent oxygen by weight, and several conceptual processes for LLOX production exist. The hydrogen reduction of ilmenite process was selected for this case study. This process is characterized as less difficult chemically and technologically than other proposed processes; however, ilmenite is only found in certain areas of the Moon. The low ilmenite content in the lunar regolith in those areas and the chemical reaction of ilmenite with hydrogen require that 300 t of lunar regolith must be mined to produce 1 t of oxygen. In addition to the mining equipment and the chemical processing plant, further support equipment is required, including regolith beneficiation equipment, oxygen liquefaction equipment, and oxygen storage tanks. Figure 3.1.5-4 shows the LLOX production plant and supporting facilities.

Supporting Systems. Several other systems support the surface operations at the lunar outpost. These include EVA systems such as space suits and airlocks, assembly and construction equipment, surface transportation elements such as pressurized and unpressurized rovers, and telecommunications, navigation, and information management (TNIM) equipment. These systems are described in detail in Volume III of this report.

3.1.5.2 Enabling Technology

To meet the objectives of this case study, several areas in planetary surface systems require further technology development. These enabling technology issues are mining and construction, mobile and stationary power, dust contamination control, extravehicular activity, lunar oxygen production, mineral beneficiation, and propellant storage and transfer. These issues are also discussed in Volume III of this report.

3.1.5.3 System Alternatives

Many studies were carried out in order to select the

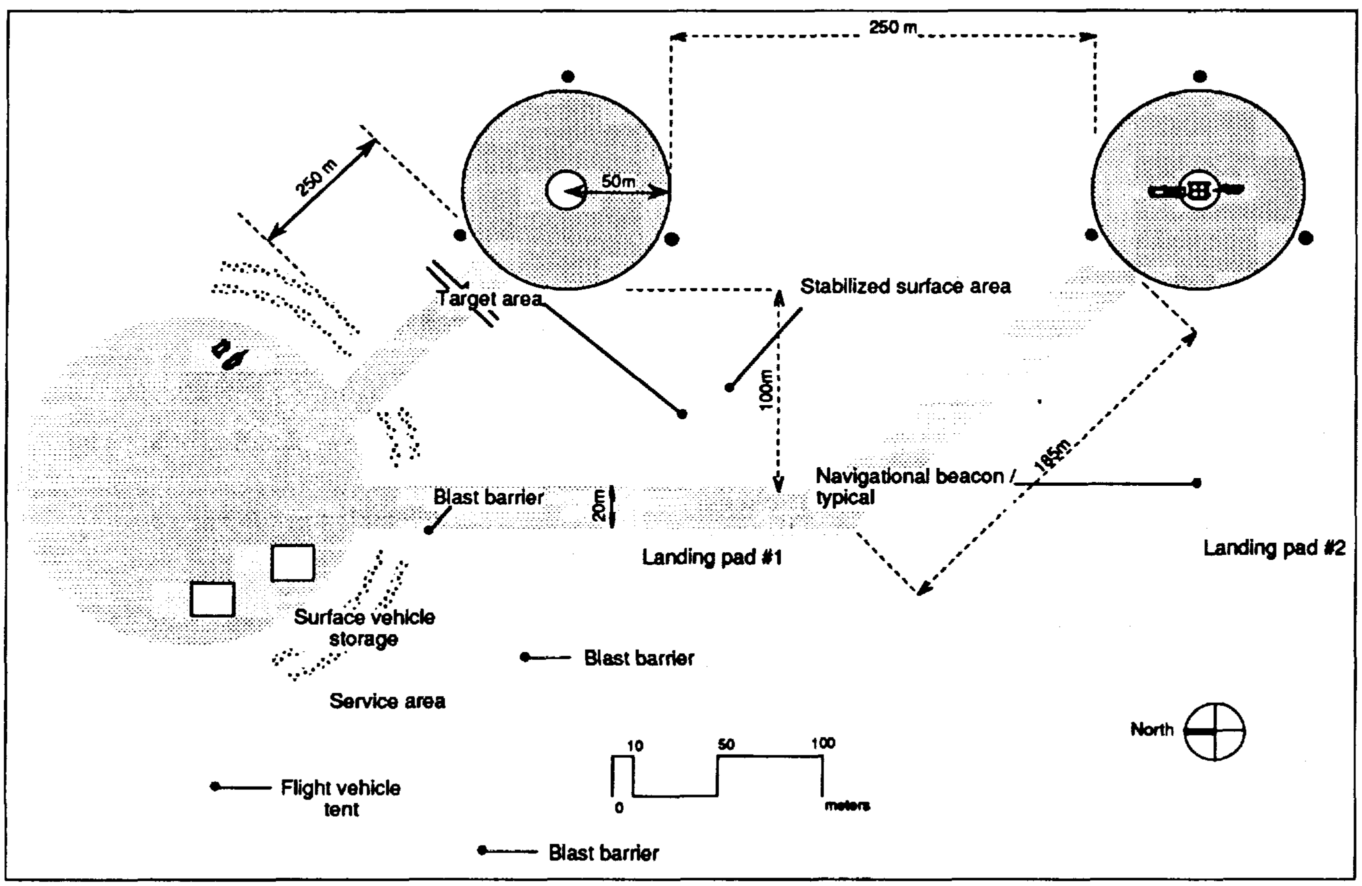


Figure 3.1.5-3.- Lunar outpost launch and landing facilities.

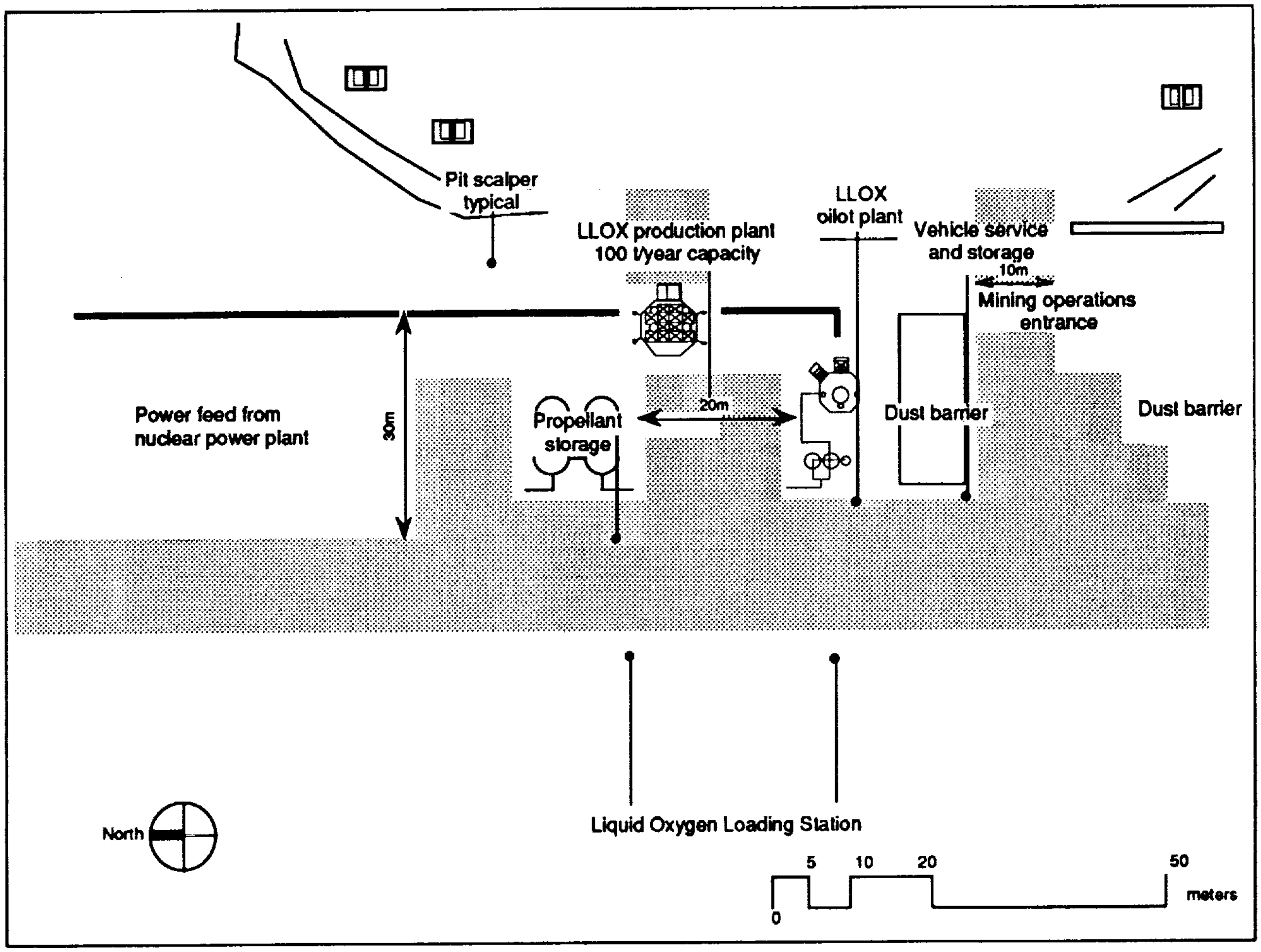


Figure 3.1.5-4.- Lunar oxygen production plant and supporting facilities.

surface systems used in this case study. Distributed Studies of Equipment and Subsystems, section 5 of Volume III in the annual report, discusses in detail the options considered and the evaluation criteria used in selecting all of the planetary surface systems.

3.1.6 Synthesized Mission Manifest

The buildup of the LEO supporting infrastructure begins in 2000 with the launch of the first phase of the Space Station Freedom growth elements. Flight tests of the lunar transfer vehicles and lunar excursion vehicles begin in 2003, with the final vehicles ready for the first mission to the Moon in late 2003. The Earth-to-lunar surface manifest for the Lunar Evolution case study for the first 13 years of the program is provided in table 3.1.6-I. The table shows the quantity of each payload and the date of its delivery to the lunar surface.

3.2 MARS EVOLUTION CASE STUDY

The objective of this case study is to emplace a permanent, largely self-sufficient outpost on the surface of Mars. Within this long-range objective are three intermediate goals: (1) establish a Mars surface outpost that grows to a permanent facility, (2) develop a significant science research capability, and (3) develop a martian moon gateway to ultimately serve as a resource for continued human expansion into the solar system.

3.2.1 Key Features

Key features of this case study include:

a. First flight for human exploration of Mars launches in 2007.

b. Vehicles are assembled in LEO at a free-flying assembly fixture.

c. Significant science research capability is provided.

d. Initial number of crew is four, increasing to seven on later flights.

e. Aerobraking is employed at Mars and at Earth.

f. Chemical propulsion is utilized throughout.

g. A martian moon gateway is developed, with fuel and oxidizer production.

h. Artificial gravity spacecraft.

TABLE 3.1.6-I.- LUNAR EVOLUTION - HARDWARE ELEMENT MANIFEST

Hardware element	Annual hardware element mass, t												
	Emplacement			Consolidation						Utilization			
	2003	2004	2005	2006	2007	2008	2009	2010	2011	2012	2013	2014	2015
Transportation systems support elements													
Lunar transfer vehicle-cargo			9.7										9.7
Lunar transfer vehicle-cargo (w/o aerobrake)	7.8												
Lunar transfer vehicle-piloted		2x18.7	18.7										18.7
Lunar excursion vehicle-cargo	3.4		3.4										
Lunar excursion vehicle-piloted		2x6.3	6.3	6.3									
Subtotal	11.2	50.0	38.1	6.3									28.4
Habitats and laboratories													
Analytical science laboratory			1.2										
Plant/animal/microbe laboratory									8.9				
Habitation module	18.8												
Airlock		3.7											
Node		4.7											
Thermal control system	2.2												
Tents		1.0	1.0										
Pressurized tunnel ramp			2.8										
Ground station			1.0										
Constructible habitat				19.6									
Constructible thermal control					6.1								
Constructible regenerative life support system					16.8								
Constructible outfitting					1.0								
Subtotal	21.0	9.4	6.0	19.6	23.9				8.9				
Facilities													
Liquefaction plant/tanks		3.0	3.0										
Fuel cell power carts		0.8	0.8										
Thermal control carts		1.2	1.2										
Navigation beacons and pad markers		0.1											
LLOX demonstration plant		0.2											
Lander facility upgrades						20.0							
LLOX plant								6.8		3x6.8			
Subtotal		5.3	5.0			20.0		6.8		20.4			
Vehicles													
Teleoperated instrumented rover			2.9										
Unmanned local rover		0.6	0.6										
Laboratory trailer			5.0										
Low power trailer			1.4										
Pressurized utility vehicle			5.0		5.0								
High power trailer					5.2								
Propellant refill vehicle									14.0				
Lunar ballistic vehicle										9.0			
Subtotal		0.6	14.9		10.2				14.0	9.0			

TABLE 3.1.6-I.- (CONCLUDED)

Hardware element	Annual hardware element mass, t												
	Emplacement			Consolidation						Utilization			
	2003	2004	2005	2006	2007	2008	2009	2010	2011	2012	2013	2014	2015
Power													
PVA/RFC (50 kW/25 kW)	10.1	10.1											
PVA (100 kW)			1.0										
SP-100 power module				5.0									
Nuclear power plant (825 kW)							20.0						
PMAD and cable (3 km)				3.0									
PMAD for LLOX plant							3x0.5	0.5					
Subtotal	10.1	10.1	1.0	8.0			21.5	0.5					
Communications													
Communications equipment		0.3	0.3										
Communication tower						0.1							
Cable (1.3 km)		1.0											
Subtotal		1.3	0.3			0.1							
Equipment													
Geologic exploration equipment		0.1								0.1			
Crane	1.9												
Digger	1.9												
Truck	1.9												
Construction hand tools		0.5											
EVA equipment		0.8											
Plant beneficiation equipment								3.4		3x3.4			
Mining equipment								4.2					
Subtotal	5.7	1.4						7.6		10.3			
Science													
Geophysical station		0.1								0.1			
Portable geophysical package		1.2								1.2			
Particles and fields instuments			0.1										
Geologic exploration instruments													
Solar observatory		1.2											
Monitoring telescope			2x0.2						0.2				
Optical telescope			0.9		0.9	0.9							
UV telescope				0.7									
IR telescope				5.7									
X-ray telescope										2.9			
Gamma-ray telescope								3.8					
Moon-Earth radio interferometer			2.1										
Earth observing instruments			2.1										
Low frequency radio array											0.2		
Biomedical laboratory instruments			3.7										
Science expedition package									0.5	0.5	0.5	0.5	0.5
Subtotal		2.5	9.3	6.4	0.9	0.9		3.8	0.7	4.7	0.7	0.5	0.5
Total annual hardware element mass	48.0	80.6	74.6	40.3	35.0	21.0	21.5	18.7	23.6	44.4	0.7	0.5	28.9

3.2.1.1 Phases of Development

The development of a permanent, self-sufficient outpost on the surface of Mars is planned to follow an evolutionary path through three mission phases: (1) emplacement, (2) consolidation, and (3) utilization. These phases are summarized in figure 3.2.1-1 and described below.

Emplacement Phase: The objectives of this phase are to accomplish the initial human landing and deploy and check out the surface habitat module. Local human exploration activities and regional semiautonomous exploration capability are initiated. The Phobos ISPP plant is delivered.

Consolidation Phase: During this phase, exploitation of local resources begins with production by the gateway propellant plant. Number of crew, Mars surface stay time, and both human and robotic exploration capabilities are expanded.

Utilization Phase: Outpost permanence is established with expanded surface facilities, greater use of Mars resources, and advanced propulsion technologies. Global exploration of Mars is demonstrated.

3.2.1.2 Mission Profile

Prior to the development of the Mars outpost, robotic missions to Mars will characterize the environment and support the selection of the outpost site. Three missions are currently defined to meet these objectives: the 1992 Mars Observer, a Mars Rover/Sample Return, and a Mars Global Network.

The mission-by-mission timeline for the case study is shown in figure 3.2.1-2. The strategy developed as part of the Lunar Evolution case study maturation process was also incorporated into the Mars Evolution case study. This common strategy begins the emplacement phase with an unmanned cargo mission, which is followed by the first piloted flight. Typical flight profiles for the cargo and personnel missions are shown in figures 3.2.1-3 and 3.2.1-4. A description of each flight is provided below.

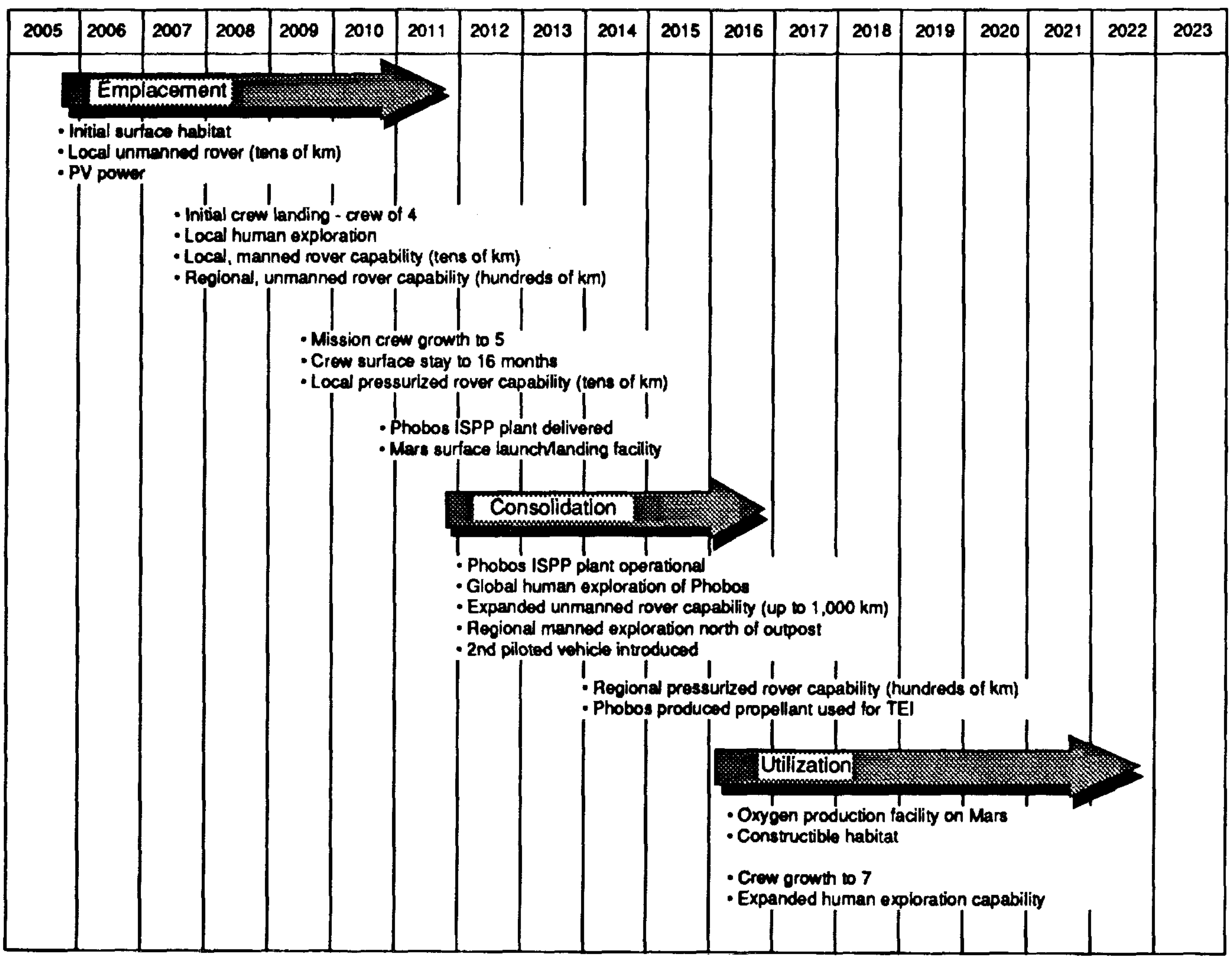


Figure 3.2.1-1.- Mars outpost development phases.

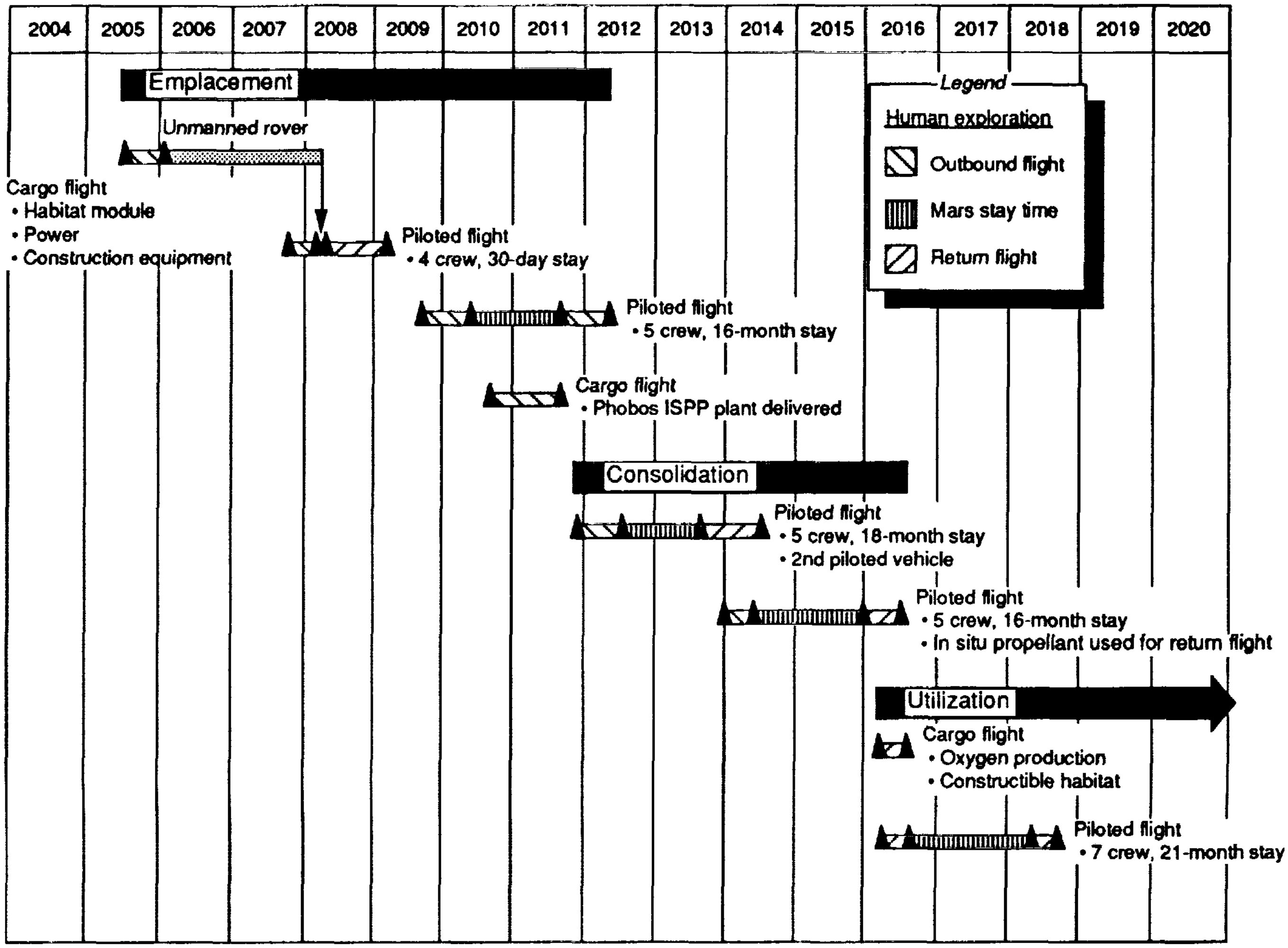


Figure 3.2.1-2.- Mars Evolution reference mission.

Flight 1 - Unmanned cargo flight: This flight departs LEO in August 2005, on a minimum energy, one-way trip. Upon arrival at Mars, the vehicle aerobrakes into a circular orbit and deploys an unmanned semiautonomous rover to the prime landing site region. Over the next several months, the rover explores the immediate region and identifies the exact landing site within the prime area. It deploys navigation beacons to assist in landing the primary cargo payload, which lands before the arrival of the crew on flight 2. The primary cargo consists of the initial surface habitat module and the construction equipment to robotically deploy the habitat. Also included in the surface payload are the necessary power supply and communication equipment.

Flight 2 - First piloted flight to Mars: This flight departs LEO in September 2007 on a 500-day round-trip class trajectory, which has the characteristic of approximately a 100-day window of stay time at Mars. Upon arrival at Mars, the piloted vehicle aerobrakes into a highly elliptical orbit (250 km x 33,120 km) at an inclination compatible with a range of potential departure declinations. The four crewmembers all descend to the surface for a nominal 30-day stay and conduct local exploration and scientific investigations in unpressurized rovers. One objective of the crew's surface stay is to verify the integrity of the habitat for subsequent crew use. A scientific payload is robotically deployed to Phobos to reconnoiter its resources and to return a sample to the piloted spacecraft for subsequent analysis to verify the use of Phobos as a propellant gateway. A robotic orbiter is also deployed to Deimos for scientific observation.

Flight 3 - First human extended duration stay at Mars: A crew of five departs LEO in October 2009 on a 1,000-day-class round-trip mission to Mars. The outbound and return transit times to Earth are minimized to the extent possible, given the propulsive capability of the piloted vehicles. The piloted vehicle aerobrakes into a highly elliptical orbit upon arrival at Mars.

The crew of five descend to the surface of Mars for approximately 500 days and live in the habitat facility emplaced and verified by the previous crew. Primary crew objectives are to demonstrate the feasibility of long-duration habitation of Mars and to conduct intensive

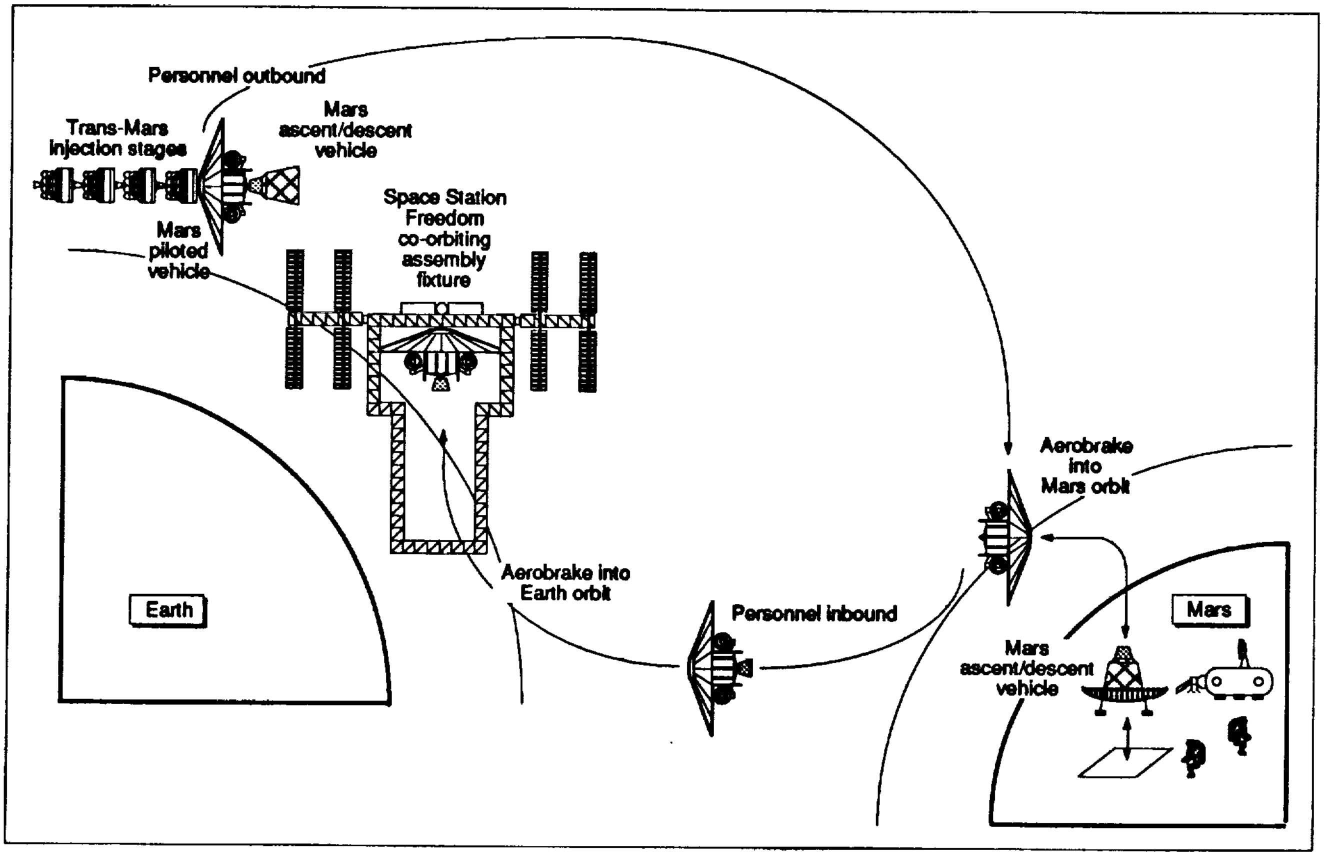


Figure 3.2.1-3.- Mars personnel flights.

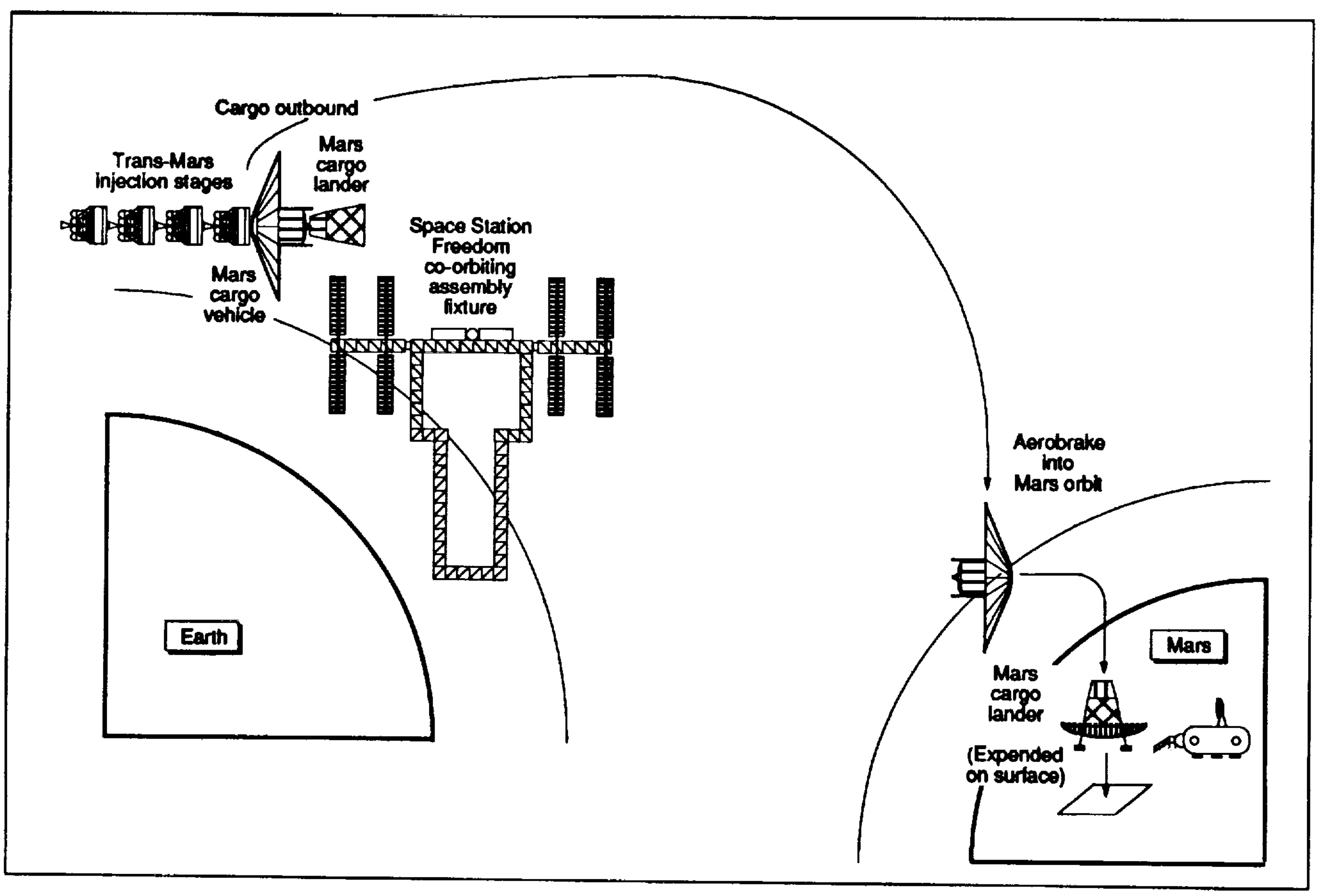


Figure 3.2.1-4.- Mars cargo flights.

regional science and exploration investigations in the vicinity of the outpost. A manned, pressurized rover is provided for an exploratory trip to scientifically interesting areas.

Flight 4: This flight is the second cargo flight, and it completes the emplacement phase. The flight launches to Mars in November 2010 using a non-minimum energy trajectory to take advantage of the Mars launch opportunity that occurs in 2010. The primary payload is the Phobos propellant production plant; therefore, the vehicle aerobrakes into a Phobos-compatible orbit, and awaits the crew arriving on the next flight. Additional supplies and equipment are also delivered to Mars on this flight.

Flight 5: This flight departs LEO in December 2011 on a 1,000-day-class round-trip to Mars. The vehicle aerobrakes into a Phobos-compatible orbit at Mars arrival, since the primary objective of this flight is the emplacement and verification of the Phobos propellant production plant. The crew also conducts global exploration of Phobos before descending to Mars for the remainder of their 18-month stay. Because this mission launches before Flight 3's crew returns, a second piloted vehicle is introduced for this flight. The Phobos propellant production plant is brought to operational status, and the crew verifies propellant production and storage; however, the propellant produced is not used until the next flight.

Flight 6: This flight concludes the consolidation phase. Launch is in January 2014, and the crew aerobrakes to Phobos upon arrival at Mars. Once the Mars transfer vehicle has been refueled for the return trip to Earth, the crew descends to the surface for approximately 500 days. The primary objective of this flight is to expand exploration and science activities on the Mars surface.

For this case study, only the first eight flights in the sequence were developed. Flights 7 and 8 begin the utilization phase and are launched in tandem during the 2016 Mars launch opportunity. Flight 7 is a cargo flight, carrying in situ resource utilization plants and a constructible habitat capable of housing 12 people.

The number of crew for Flight 8 is increased to seven to take advantage of the scientific opportunities allowed by the expanded habitat facilities. Their primary objective is the erection of the constructible habitat. Surface resource utilization and use of Phobos propellant allows expanded exploration.

Possible mission types examined for this case study range from "all-up" configurations, which send a mix of cargo and crew on a single flight, to split missions, which send cargo and crew on separate flights. Trajectories employed for this year's studies fall into two categories grouped by round-trip flight time: (1) 500-day class missions (also known as opposition-class) and (2) 1,000-day class missions (also known as conjunction-class). All piloted trans-Mars trajectories are characterized by having a free return capability to Earth. That is, following trans-Mars injection, no propulsive maneuvers are nominally required to return to Earth in the event that the mission is aborted prior to Mars orbit capture. The initial piloted flight is an all-up configuration using a 500-day class trajectory. Later flights use 1,000-day class trajectories. Vehicles are sized to accommodate early flight requirements and then are reused to accomplish, to the maximum extent, more aggressive later missions.

The trajectory data for the Mars Evolution case study are shown in table 3.2.1-I. The key assumptions and requirements that were imposed upon the trajectory generation process, and that are implicit in the trajectories listed in the table, are:

a. A free-return capability was required for all the piloted Mars trajectories; i.e., the piloted vehicle could return to Earth without any propulsive maneuvers following trans-Mars injection at Earth departure.

b. Atmospheric entry velocities (for aerobraking) were limited to 9,500 m/sec at Mars and 13,500 m/sec at Earth.

c. Fast transit times (outbound and inbound) consistent with the SRD ΔV budgets were used.

3.2.2 Science Opportunities and Strategy

The science program described here has not received widespread scrutiny and approval by the scientific community. It is intended to reflect a basic, preliminary approach to human Mars exploration within the specific framework and mission characteristics of the Mars Evolution case study. The science program must still evolve through the normal maturation process of definition, refinement, review, and consensus within the science community. However, some benefit was derived from early efforts of various science strategy and oversight groups recently established and sponsored by OEXP. Versions of the MASE science program were made available at different stages for comments and input to OEXP, the Exploration Science Working Group (EXSWG), the Center Exploration Programs Scientists (CEPS), the Opportunity Analysis Group, and the newly formed Mars Science Strategy Group at Ames Research Center.

The appropriate balance between manned and unmanned exploration warrants further examination. The science developed for this case study assumes a particu-

TABLE 3.2.1-I.- MARS EVOLUTION CASE STUDY TRAJECTORIES

Flight	Launch year	Launch date	ΔV TMI m/sec	Arrival date	Outbound flight time, days	Mars stay time, days	Departure date	ΔV TEI m/sec	Return date	Return flight time, days	Total mission duration, days
1.	2005	8/11/05	4,210	1/16/06	158	-	Cargo	-	-	-	-
2.	2007	9/26/07	4,500	2/25/08	152	0	Abort	0	10/05/09	588	740
	2007	9/26/07	4,500	2/25/08	152	30	3/26/08	2,500[1]	2/17/09	328	510
3.	2009	10/22/09	4,150	5/14/10	204	0	Abort	0	12/7/12	939	1,143
	2009	10/22/09	4,150	5/14/10	204	501	10/3/11	1,250[1]	4/20/12	200	911
4.	2010	11/25/10	4,600	9/17/11	296	-	Cargo	-	-	-	-
5.	2011	12/20/11	4,400	5/27/12	159	0	Abort	0	1/15/15	963	1,122
	2011	12/20/11	4,400	5/27/12	159	529	11/7/13	1,700[2]	5/26/14	200	888
6.	2014	1/11/14	4,010	6/20/14	160	0	Abort	0	2/9/17	966	1,126
	2014	1/11/14	4,010	6/20/14	160	564	1/4/16	1,700[2]	7/7/16	185	909
7.	2016	2/28/16	3,800	7/27/16	150	-	Cargo	-	-	-	-
8.	2016	3/12/16	4,100	7/20/16	130	0	Abort	0	3/20/19	973	1,103
	2016	3/12/16	4,100	7/20/16	130	639	4/20/18	2,200[2]	8/28/18	130	899

(1) Depart from 250 km x 1 sol orbit.
(2) Depart from 250 km x 9,378 km (Phobos) orbit.

lar mix of the two, but future scenarios need to seriously consider what an optimal mix would actually be.

3.2.2.1 Opportunities

Mars provides opportunities for human exploration that are markedly different from those of the Moon because Mars has water-related features and a dynamic atmosphere. Although surface conditions today cannot support liquid water, the relict runoff channels and valleys indicate that the climate of Mars was dramatically different 3.8 billion years ago; i.e., the time when the erosional valley networks were estimated to be formed. Exploration will surely include the search for water in the atmosphere, polar caps, subsurface and surface permafrost, and possible subsurface aquifers.

The martian atmosphere is high in CO_2, and it undergoes seasonal changes. Weather on Mars also is seasonal. Local to global dust storms originating in the southern hemisphere near perihelion and the southern summer solstice vary in strength, duration, and dust density. These phenomena prompt a pursuit of atmospheric and weather studies.

A martian outpost also offers a variety of unique scientific opportunities that can be accomplished on the surface to study the physical properties and processes of the planet. These opportunities include studying martian history, morphological development, origin, resource potential, and natural environmental characteristics.

Potential site investigations span the science disciplines and include studying geological and geophysical characteristics. This includes collecting samples to study rock, soil, and regolith compositions, distributions and ages (dating analysis done in Earth-based laboratories), and investigating subsurface properties, hydrology, stratigraphy, erosion phenomena, dust mobility, and regolith maturity.

The possibility of past or extant life on Mars stimulates exobiological studies, including the search for the oldest exposed sedimentary deposits, associated fossils, organic compounds, and biogenic elements. Relict lake beds are also desirable areas to search for fossil life-forms.

Traverse science provides opportunities for regional mapping of geologic formations and geomorphology, geophysical surveys of structure, magnetic fields, gravity, and resource identification and distribution. Regional traverses would also study regional volcanics and the location and form of surface and subsurface water, and would validate and calibrate remotely sensed Mars orbital data.

Surface geophysical monitoring stations measure seismicity, heat flow, magnetic fields, gravity, atmospheric gases, and meteorological activity, and monitor wind, clouds, and dust storms, and dust density and composition.

In a Mars laboratory, geochemical analyses could be conducted of surface materials in support of resource

evaluation as well as basic research on martian materials. Life sciences experiments in a Mars surface laboratory include exobiology; experiments on plant growth using hydroponics and martian soil; animal and human performance, behavior, and physiology in the Mars environment; and the long-term biological effects of radiation and dust in the planetary environment.

Since the travel time between Earth and Mars is long, significant science investigation, referred to here as "cruise science," would be done en route. In particular, studies of human physiology/psychology and human responses to radiation and the microgravity environment would be conducted. A full suite of particles and fields experiments would also be possible, including observations of the solar wind, the Sun, cosmic rays, and gamma-ray and X-ray bursts. Astronomical observations could be made at optical, IR, and UV wavelengths. It is assumed that similar science experiments will be conducted on all Mars flights, with the human adaptation experiments limited to piloted flights.

A more detailed treatment of Mars science opportunities is included in Volume VI of the FY 1989 Exploration Studies Technical Report.

3.2.2.2 Strategy

The general science strategy includes the following points:

a. Mars exploration capability expands from local (unpressurized manned and unmanned) to regional (pressurized manned and unpressurized umanned) to global scales.

b. Mars science will be progressively enhanced, building toward understanding the planet's past and present.

c. Overall scientific capability will progressively grow at the Mars outpost, partly as a function of expanding pressurized space available for scientific experiments. Analytical geochemistry/petrology capabilities will expand along with plant/animal/microbial experiments and human adaptation studies.

As part of this strategy, there will be a mix of experiments that require human real-time operation (e.g., active seismic experiments, life sciences laboratory work) and observations by instruments that are deployed either manually or robotically, or that monitor martian phenomena without human tending (e.g., geophysical monitoring). The latter can be left behind between piloted flights to transmit data back to Earth.

A network of multi-instrumented geophysical/meteorological stations is emplaced on Mars to enable global geophysical and atmospheric observations. This network uses both manually deployed stations and stations emplaced on penetrators or unmanned soft landers, located far from human access. Each station measures physical properties such as long- and short-period seismicity, heat flow, temperature, humidity, wind speed and direction, atmospheric volatiles, atmospheric chemistry, and dust obscurity.

The specific science strategies and objectives associated with the three phases of the case study are summarized in figure 3.2.2-1. Each phase of the case study includes a science payload to implement the science program. The science mass for these payloads (including cruise science) is accommodated in its entirety in the Mars Evolution manifest. A detailed description of the science program for the first seven flights with associated science instrumentation is included in Volume VI.

Emplacement Phase. The science objectives for the Mars emplacement phase are:

a. Conduct unmanned robotic site evaluation and exploration of the Mars landing vicinity prior to piloted landing.

b. Conduct local manned exploration near the Mars outpost, including geology, geophysics, composition and age (dating done on Earth) of rocks and sediments; study stratigraphy; search for life/fossils; study surface and subsurface structure and hydrology; and observe atmospheric chemistry from balloons and rockets. In general, the strategy is for the unmanned rover to do "path planning" and sample collection in preparation for manned expeditions.

c. Locate and evaluate local resources (water, minerals).

d. Begin a network of geophysical/atmospheric science stations that are both manually and robotically deployed.

e. Conduct unmanned exploration of Phobos and Deimos.

Consolidation Phase. During this phase, Mars stay time is lengthened, and the range of the pressurized rover is extended to "regional" (up to 1,000 km). Local human and regional unmanned rover exploration are augmented by regional human traverses for geologic and geophysical exploration. The science objectives change with the expanded capabilities on Mars. For this phase, objectives include:

a. Extend human exploration capabilities up to 1,000 km, study geology and geophysics, search for life/fossils; continue unmanned exploration, including sample selection and collection and path planning.

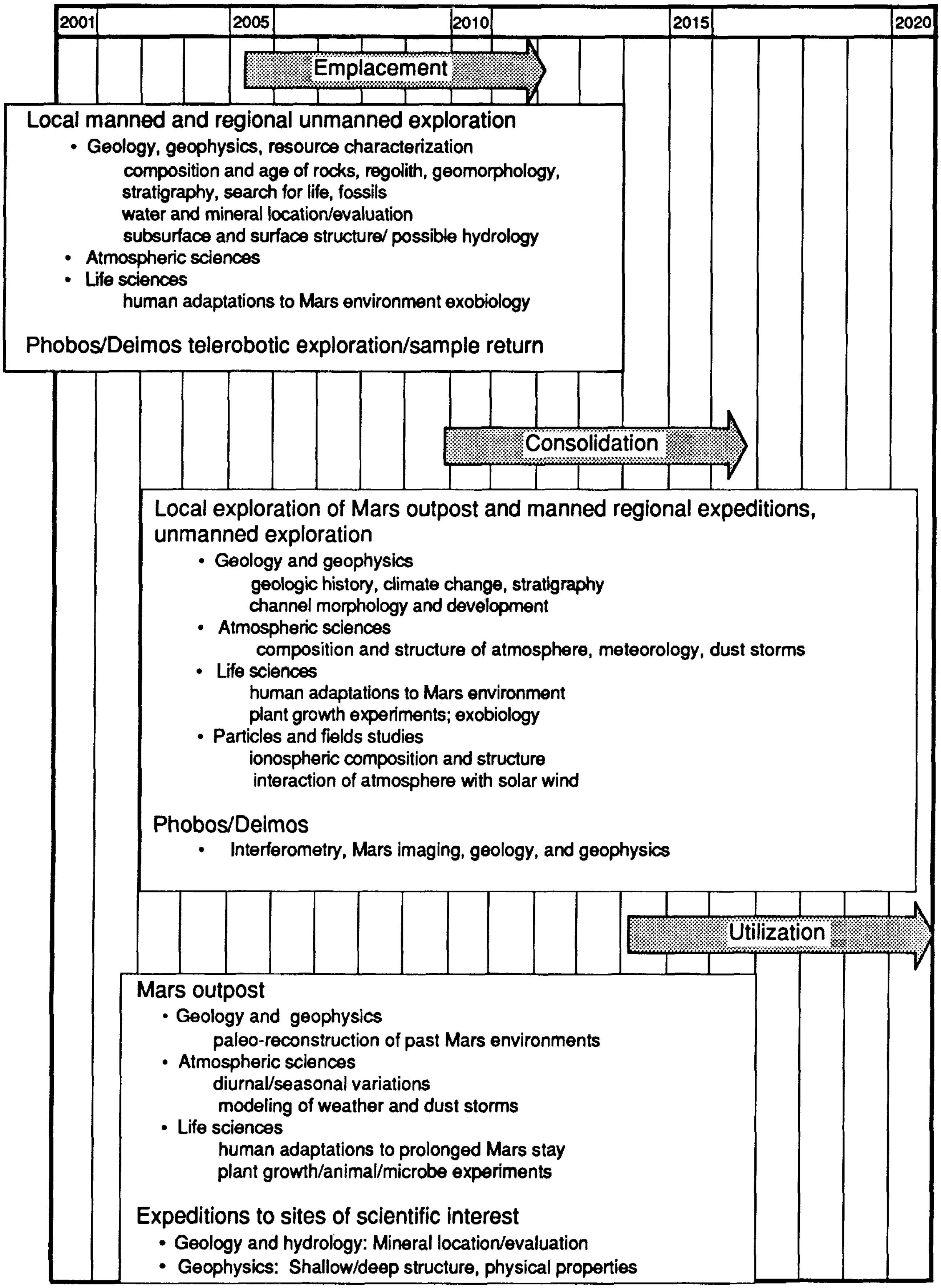


Figure 3.2.2-1.- Mars Evolution case study: science strategy and objectives.

b. Study regional geomorphology, channel morphologies, and development.

c. Supplement analytical laboratory capabilities for experimentation with resource utilization and materials processing; begin plant growth experiments and human adaptation research.

d. Expand geophysical/atmospheric science network and balloon coverage to extend understanding of martian weather, dust storms, and geophysical properties.

e. Conduct a manned reconnaissance science expedition to Phobos, partly in support of Phobos resource development.

Utilization Phase. The utilization phase continues to use the Mars outpost to perform science activities, with an emphasis on the expanded life sciences effort in support of plant growth experiments, experiments with microbes and animals, and additional work with human adaptation. The strategy also includes extended regional traverses to the northern regions to continue to search for water.

The expanded network of global geophysical and atmospheric sciences monitoring stations has sufficient areal and temporal coverage to enable modeling of planetary trends in weather, dust storms, seismicity, subsurface structure, and other geophysical properties. The latter part of the utilization phase marks the first opportunity to attempt comprehensive paleoreconstruction of planetary processes in Earth-based universities and laboratories.

During the utilization phase, more distant expeditions allow geology and resource exploration at sites beyond the range of the regional manned rover. One candidate site for a manned ballistic-type mission is one of the Mars polar areas to search for the possible existence of water. Another site might be the volcanic province of Tharsus. The typical science payload for such missions would include an unpressurized rover equipped with an electromagnetic sounder, geologic and geophysical exploration equipment, and a geophysical/atmospheric sciences station. Specific science strategies for these missions would depend upon the particular site.

Cruise Science. The science strategy for "in-flight" science is:

a. Piloted flights will conduct a suite of experiments on crew studying human response and adaptation to zero-gravity environment.

b. Piloted and cargo flights will operate astronomy (optical, UV, IR) and solar telescopes and particles and fields experiments.

The science payload includes optical, UV, IR, and solar telescopes; gamma-ray and X-ray burst detectors; particles and fields instrumentation; and biomedical instrumentation.

3.2.2.3 Surface Science Scenario

This discussion is intended as an example of how early flights in the Mars Evolution case study could reasonably be executed on the Mars surface. As such, the description provides some operational scope and reality to the science mission, as it is currently configured, in terms of science objectives, stay time, EVA limits, crew time, and surface vehicle availability. The landing site is a starting point from which operations can commence in order to scope the activity. Although the chosen site is a valid candidate, it was selected as typical and does not imply advocacy.

The landing site for this case study is located on the Mars equator at 33.5 degrees west longitude in an area generally known as the Chryse Basin complex. This area is a wide fluvial plain, approximately 56 km in width, resulting most likely from a combination of water outflow from the Valles Marineris complex and melting ground ice; the latter effect caused the area to subside. Canyon walls along the flood plain are 3 to 5 km high. Hills on the plain loom approximately 1 km high.

Flight 1 is a cargo flight that includes a teleoperated (from Earth) unmanned rover as part of its payload. The rover is delivered to Mars from the orbiting cargo vehicle and, once on the surface, begins a survey of the landing site area. When the site survey is completed, the cargo vehicle lands at the Mars outpost site. The teleoperated rover is then dispatched to the 3 to 5 km high wall about 100 km away. The rover collects samples and performs geophysical measurements during the traverse to and from the wall, taking between 25 to 35 days total travel time. The rover photographs the canyon wall strata both in the visible and infrared portions of the electromagnetic spectrum. When the rover returns to the outpost, the geological samples collected are deposited at the landing site with the cargo vehicle for examination by crewmembers on later flights.

After depositing the samples at the outpost, the teleoperated rover proceeds southward down the fluvial plain and then westward into the Ganges Chasma to the canyon origin, and finally back to the outpost. The trip will cover approximately 2,700 km and take from 200 to 300 days to complete. During the traverse, samples are selected, geophysical measurements are taken, and the canyon walls are photographed in the visible and infrared. The geological samples are returned to the Mars outpost for examination upon the arrival of the crew from Flight 2.

Flight 2, with a crew of four, deploys a teleoperated rover approximately 400 km north of the outpost in a portion of the Chryse Basin complex flood plain known as Simud Vallis, which connects to the outpost area. The rover arrives at the outpost 30 days later, in time to deliver the samples it has collected to the crew before they depart.

During the 20- to 30-day surface stay time, two crewmembers, each in a separate unpressurized rover, deploy a set of geophysical and atmospheric measuring instruments and conduct a series of traverses collecting soil and rock samples. A map of the traverses in the near vicinity (20 km) of the Mars outpost is shown in figure 3.2.2-2.

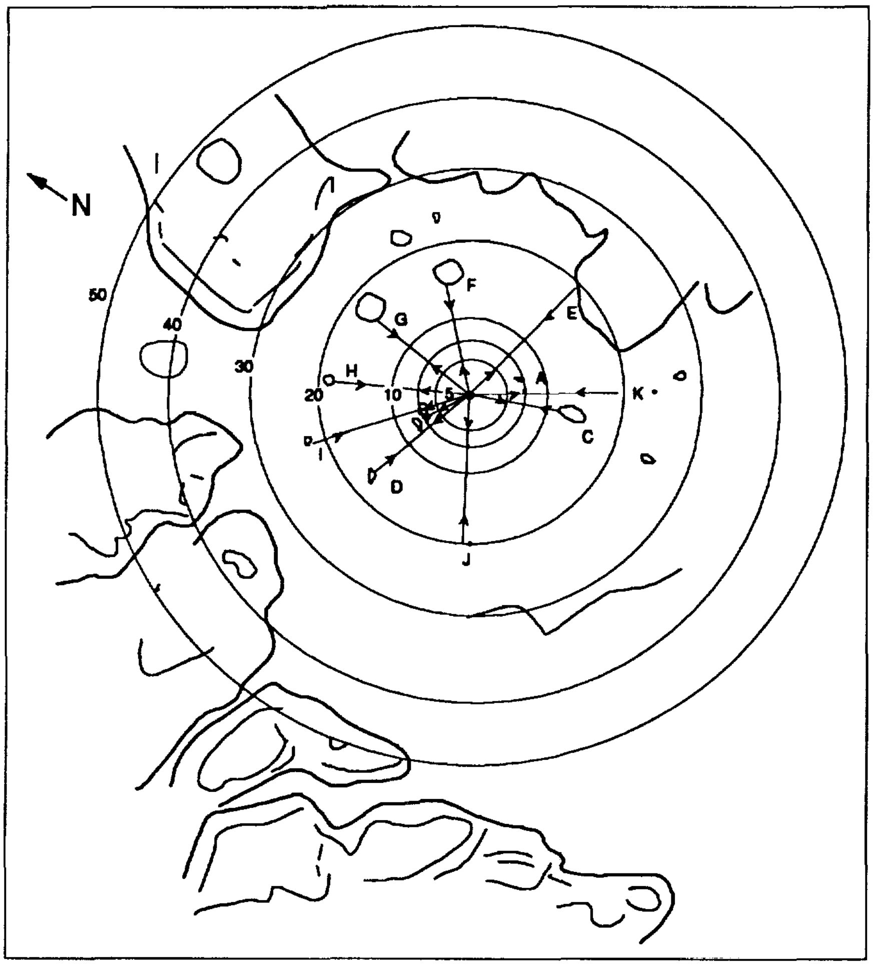


Figure 3.2.2-2.- Geologic traverses by crewmembers with unpressurized rovers within 20 km radius of Mars outpost.

3.2.3 Transportation Systems

The transportation system presented here is the MASE synthesis of the implementation provided by the Transportation IA. The information represents some modification of the information presented in Section 3 of Volume II of this report. These modifications are briefly discussed in Section 5 of that volume. For ETO transportation elements, the STS is constrained to two flights per year beyond those required for Space Station Freedom operations. An HLLV with a 12.5 m diameter payload shroud and a payload capability of 140 t to Freedom orbit are assumed. For space transfer vehicles, the Mars transfer vehicle (MTV) is designed to be reusable without major maintenance operations. Due to the uncertainty of aerobrake reusability, the aerobrake, which is used both at Mars and for crew return to Freedom, is jettisoned following Earth entry.

3.2.3.1 Elements and Systems

ETO Vehicles

Space Transportation System (STS): capable of transporting crew of four and cargo of 10 t from Earth to LEO.

Heavy-lift launch vehicle (HLLV): a vehicle capable of lifting 140 t from Earth's surface to LEO, assumed to be available for a maximum of four flights per year. Several design options were considered for an HLLV of the 140 t lift class. These are shown in Section 4.1 of this document. The Shuttle-Z concept was assumed for manifesting purposes for the Mars Evolution case study. The Shuttle-Z concept burns the trans-Mars injection (TMI) stage as the HLLV third stage, and this stage arrives in orbit dry. The payload lift capability to LEO (in addition to the dry TMI stage) is 140 t.

Space Transfer Vehicles

Mars transfer vehicle (MTV): a generic term for the vehicle that transports crew and cargo from LEO to Mars orbit. The piloted configuration is referred to as the Mars piloted vehicle (MPV). The cargo configuration is referred to as the Mars cargo vehicle (MCV).

Mars excursion vehicle (MEV): a generic term for the vehicle that transports crew and cargo from Mars orbit to the surface of Mars or a martian moon. The piloted configuration is referred to as the Mars descent vehicle (MDV). The cargo configuration is referred to as the Mars cargo lander (MCL).

Piloted Vehicles

Mars piloted vehicle (MPV): a piloted vehicle for a three-, five-, or seven- crewmember mission that uses

chemical-H_2/O_2 propulsion/aerobraking to provide transport between LEO and Mars orbit. The vehicle has a five-deck vertically oriented artificial-g facility that provides a range of gravity conditions. This vehicle can be reusable with LEO refurbishment after each flight.

Earth crew capture vehicle (ECCV): a piloted vehicle for direct entry of five crewmembers (increases to seven in 2014) from the returned MPV to Earth's surface with a 200 kg cargo. The vehicle consists of a crew cab and a heat shield. Due to the high entry velocities associated with the 500-day class trajectory (Flight 2), direct entry to Earth by the crew in the ECCV is an option. However, the baseline is that the entire MPV is returned to a Space Station Freedom-compatible orbit for servicing and reuse on Flight 3. For the 1,000-day class missions, it was assumed that the entire MPV is aerobraked back into a Space Station Freedom-compatible orbit. The ECCV serves as a lifeboat, in case of MPV malfunction, for these cases.

Mars descent vehicle (MDV): a piloted vehicle for a five-crewmember (increases to seven in 2004) mission from Mars orbit to the Mars surface. The vehicle carries the Mars ascent vehicle plus 25 t of cargo. The vehicle components include a crew cab, a Mars ascent vehicle (MAV) and a lander/aerobrake propulsion module.

Mars ascent vehicle (MAV): a piloted vehicle for five crewmembers (increases to seven in 2004) and 200 kg cargo transport from the Mars surface to Mars orbit. It consists of a crew cab and an ascent and orbit transfer propulsion module. Ascent propellant is storable bipropellant.

Cargo Vehicles

Mars cargo vehicle (MCV): an expendable, unmanned vehicle designed to deliver 187 t payload from LEO to Mars orbit. The propulsion system uses chemical H_2/O_2 propellants for impulsive engines and uses storable propellants for the reaction control system (RCS).

Mars cargo lander (MCL): an expendable, unmanned vehicle designed to transport 50 t cargo from Mars orbit to the Mars surface. The vehicle uses the same MDV lander/aerobrake propulsion module.

Figure 3.2.3-1 shows the piloted and cargo configurations of the Mars transfer and excursion vehicles. The MEV is expendable and transports crew and cargo from Mars orbit to the surface of Mars or its moons. Advanced

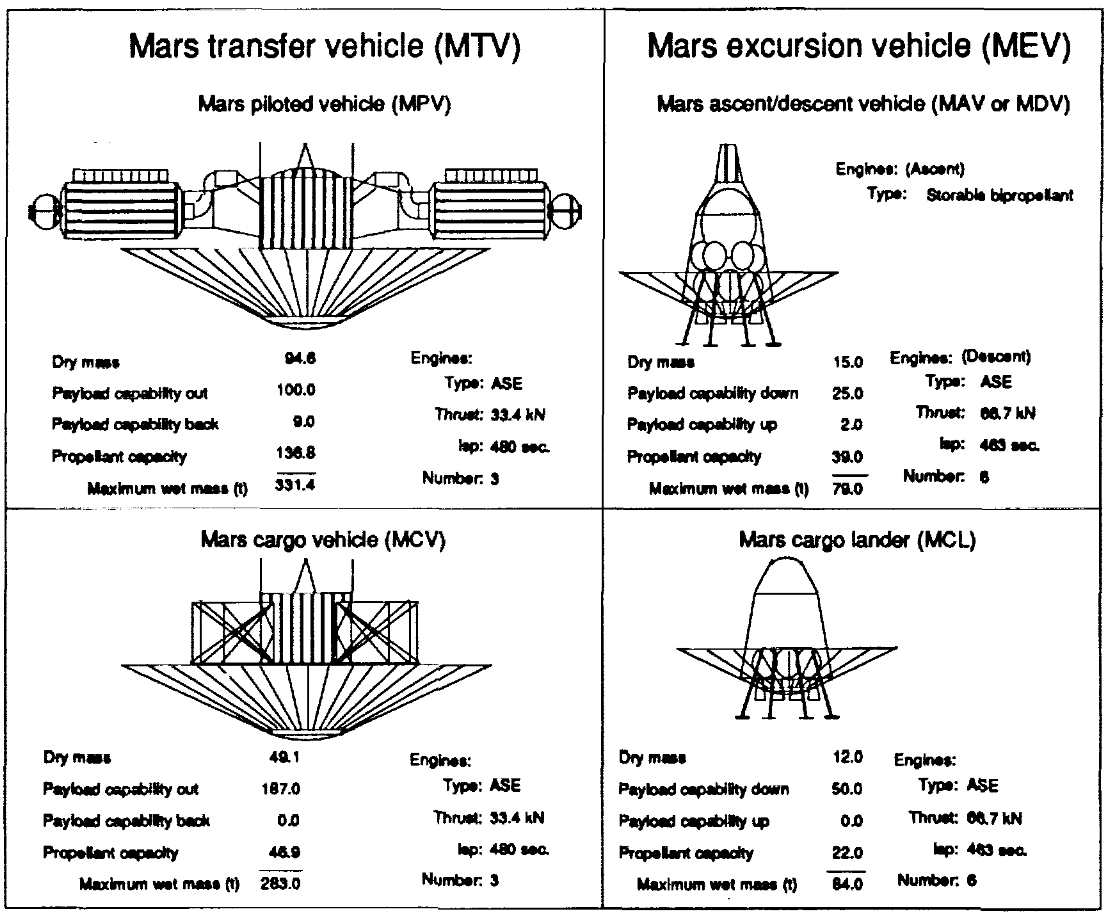


Figure 3.2.3-1.- Mars Evolution MTV and MEV configurations.

Space Engines are used on the Mars transportation vehicles, with the exception of the ascent engines of the piloted MEV, which uses a storable bipropellant.

Trans-Mars injection stage (TMIS): this propulsion module is used to transport vehicles from Earth orbit to Mars orbit. It uses chemical $H_2/0_2$ propulsion and contains one engine per module/stage. Each engine provides 2,372 kN of thrust at a specific impulse of 471 sec. Three TMI stages are required for cargo flights, and four TMI stages are required for piloted flights.

Trans-Earth injection stage (TEIS): this propulsion module is used to transport vehicles from Mars orbit to Earth orbit. It uses chemical $H_2/0_2$ propulsion. Each engine has a thrust of 89 kN at a specific impulse of 480 sec.

3.2.3.2 Enabling Technology

A major enabling technology for Mars Evolution transportation systems is on-orbit assembly capability. This capability includes coordinated large-scale EVA/teleoperated integration for assembling the aerobrake and the Mars transfer vehicles, and fluid connects for cryogenic and ambient temperature feed lines to allow in-space rapid cryogenic propellant transfer (TMIS needs 10 to 50 t/hr). In addition to on-orbit assembly, vehicles and vehicle systems need 10- to 15-year nominal lifetimes to be effective. Reusable descent/ascent cryogenic engines are also required and must be dust-tolerant and throttlable. In the area of vehicle automation, two other significant technologies that are enabling for the Mars Evolution case study are autonomous landing and autonomous rendezvous and docking capabilities required for the cargo flights. In the area of crew health and safety, two key areas requiring enabling breakthroughs are radiation protection and a resolution on the question of artificial-g versus zero-g vehicles.

3.2.3.3 System Alternatives

In addition to the reference set of vehicles, nuclear thermal rockets (NTR) were studied to determine their effects on the mission. NTR missions assumed the use of aerobraking and terrestrial propellants only. Performance of NTR using Phobos propellants for return would yield reduced flight times and increased payload capability over missions using only terrestrial propellants. Aerobraking for the NTR alternative revealed an increase in the payload capacity of an NTR Mars transfer vehicle by about a factor of three. For the aerobrake assumptions used in the Mars Evolution case study (single 26 t aerobrake used at both Mars and Earth), an all-propulsive NTR mission does not show an IMLEO advantage over the aerobraked chemical baseline. With comparable performance, however, the NTR provides a credible "all-propulsive" option.

3.2.4 Orbital Node Systems

The SRD stipulated that the assembly of Mars vehicles would occur on a free-flying man-tended assembly fixture co-orbiting with Space Station Freedom. The node must support: (1) vehicle mating/assembly and demating/disassembly, (2) space construction of elements of STVs, (3) deployment and retrieval of transfer vehicles, and (4) element and integrated vehicle on-orbit checkout. Communications must be provided between the assembly node and Earth stations.

3.2.4.1 Elements and Systems

The Mars transfer vehicle is not assembled on Space Station Freedom, but is accommodated on a free-flying, co-orbiting assembly fixture. Freedom's main requirements are to house the transient mission crew and Mars transfer vehicle (MTV) assembly crew, support the advanced development of Mars outpost and MTV systems, and provide a life sciences research capability. The Space Station Freedom evolution growth deltas required to support the mission requirements are presented in table 3.2.4-I. The programmatic schedules and further details are provided in section 4.2 of this document.

TABLE 3.2.4-I.- MARS EVOLUTION CASE STUDY SPACE STATION FREEDOM GROWTH DELTAS

Δ1	Two 25-kW solar dynamic modules; two 25-meter transverse boom extensions; space-based OMV and space-based OMV accommodations
Δ2	Upper/lower keels and booms
Δ3	One habitat module; two resource nodes
Δ4	Two 25-kW solar dynamic modules; servicing facility phase 1
Δ5	One large pocket laboratory (artificial-g); one large pocket laboratory (CELSS); servicing facility phase 2
Δ6	Life sciences laboratory module; two resource nodes
Δ7	Phase 3 servicing facility (completed Customer Service Facility)
Δ8	One large pocket laboratory (quarantine facility)

The characteristics of the orbital node for the Mars Evolution case study are:

a. The node will be a free-flying fixture with accommodations for one Mars transfer vehicle and cargo.
b. The node should be gravity gradient stable with two planes of symmetry.
c. Assembly activities should be located near the node's center of mass.
d. Payload transfer distances should be minimized.
e. Separate docking facilities for crew and payload deliveries should be provided.
f. Power collecting devices should be staggered to minimize mass and area property fluctuations.
g. Sufficient structure should be provided to ensure structural integrity and controllability during all phases of assembly and operations.
h. The node should be designed for safe manned operation.

A configuration for this node, termed the Skyhook transportation node configuration, is shown in figure 3.2.4-1. Elements consist of:

a. Two Space Station Freedom-type pressurized habitation modules that provide volume and redundancy necessary for man-tended operations
b. One hyperbaric airlock used to transfer crew to and from the node
c. Eight photovoltaic solar panels and four thermal radiators, which provide 75 kW used for the pressurized modules, logistics, avionics, assembly, equipment, and thermal control
d. 88 bays of truss
e. Two alpha joints
f. Four beta joint motor boxes
g. One OMV docking ring
h. Eight RCS jet thruster clusters

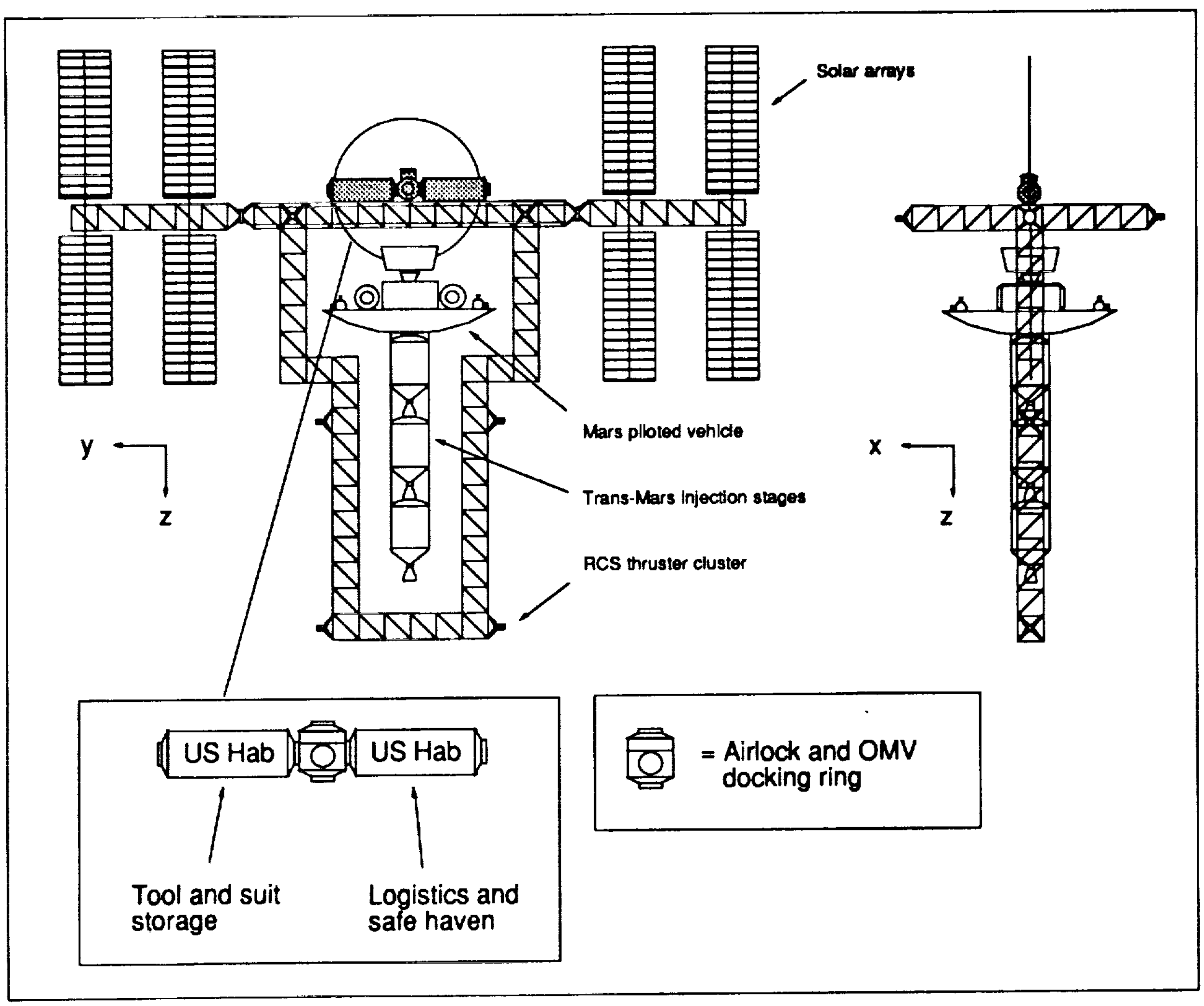


Figure 3.2.4-1.- Skyhook transportation node configuration for Mars vehicle assembly.

The assembly node can be carried to its 500 km, 28.5 degree inclination orbit by the Shuttle, Shuttle-C, or other HLLV. Using the Shuttle requires 11 flights to assemble the node. More capable vehicles require fewer flights. When comparing ETO capabilities, the Shuttle configuration payloads are restricted by mass, whereas the Shuttle-C configuration payloads are restricted by volume.

3.2.4.2 Enabling Technology

Several technologies would be needed to support assembly activities during operation of the node. To support the fueling of STVs at the node, technologies for propellant transfer and long-term storage are needed. An automated fault detection/isolation and check-out system is needed to verify capabilities of vehicles and vehicle systems received by and assembled at the node. Technologies that enable STV docking and berthing and interfaces for mating the node with the STV/payload must be developed. In addition, technologies for the support of the assembly facility and equipment must be provided.

3.2.4.3 System Alternatives

Alternatives considered included the location of the node, the degree of manned capability provided, power system selection, and RCS configuration. For this study, the node was considered to be a free-flyer located in the 500 km, 28.5 degree inclination Freedom orbit. Because of the 26-month window for Mars missions, the node was assumed to be man-tended to reduce logistics for crew support.

3.2.5 Planetary Surface Systems

Planetary surface systems are designed to establish a Mars outpost through three phases of development: (1) emplacement, (2) consolidation, and (3) utilization. In the emplacement phase, planetary surface elements include a habitat and vehicle support for a crew of four. In the consolidation phase, a propellant plant on Phobos is established, the Mars surface launch and landing facility is completed, and a crew of five is supported. During the utilization phase, in situ resources are utilized, a LOX plant on the Mars surface is completed, regional pressurized rover capability for hundreds of km traverses is provided, and a crew of seven is supported.

3.2.5.1 Elements and Systems

Habitation. The first crew to the Mars surface will initially reside in the lander. The first habitat to be emplaced is a Space Station Freedom-derived common module designed to support a crew of five. It contains sleeping berths, a galley, a health maintenance facility, and limited work and storage space. All elements are self contained except the power supply, which is externally located. This minimizes set-up activities, requiring the first crew to only connect the external power supply and certify the module for pressurized occupation.

To accommodate the growing demands on the outpost, a 16-meter diameter constructible habitat was chosen for the second generation of outpost habitation. The reference concept is an inflatable structure with multiple assembled floors.

EVA Systems. EVA will be required for a variety of operations, including construction, operation, and maintenance of outpost systems, as well as for performing scientific exploration of Mars. The suit pressurization system is baselined at 8.7 psi. EVA crews will also be supplied with Apollo- and Shuttle-type hand tools and unpressurized rovers for local area access.

Life Support Systems. The life support system selected for the evolutionary martian outpost is a regenerable physical-chemical system. This system evolves in two phases, initial and advanced, as the outpost expands to include larger crews. The initial system, a derivation of the Space Station Freedom system, is integrated into the initial module, accommodating a crew of five. With the construction of the inflatable habitat, an advanced life support system is installed to support crews extending to seven persons. The advanced system marks improvements over the initial system by advancing the recovery of materials from solid wastes and reducing resupply of system expendables.

Thermal Control. Thermal control systems (TCS) for the initial habitation module and constructible habitat require active as opposed to passive measures. The TCS must actively acquire internal equipment and metabolic heat loads and reject these loads to the external environment. Heat pumps, required on the Moon, can be avoided on Mars due to the cold environment and the short martian day. Unique martian parameters critical to TCS design include extreme changes in surface and atmospheric temperatures, changing winds, and degradation of surface properties. For radiative purposes, the martian environment is a good heat dump, and a horizontal radiator may be used. If dust contamination is a concern, vertical radiators with appropriate wind shields or square radiators oriented parallel to the prevailing wind direction may be used.

Radiation Protection. Surface crews will need protection from the constant galactic cosmic ray (GCR) background and infrequent but life-threatening solar flare events. The martian atmosphere, though thin, provides excellent protection except at the zenith. Recent studies indi-

cate that for crew stay times of less than 1 year, no GCR protection is required. Longer stay times will require GCR protection at some level. Solar flares can be treated on a contingency basis due to their short duration; a temporary shielded shelter should be adequate. However, once full GCR protection is provided, protection is automatically provided against solar flares.

Energy. Studies have indicated that an evolutionary outpost can be initiated with a 50 kWe power system. A photovoltaic array (PVA)/regenerative fuel cell (RFC) has been baselined to fulfill this need. The array provides daytime power for the outpost and for electrolyzing water into fuel and oxidizer for the fuel cell, which provides power during the night. The power system is robotically emplaced and deployed before the first piloted mission. As the outpost evolves, and power demands increase above the 100 to 200 kWe range, a nuclear-based power system becomes an attractive option from a mass perspective. The baselined system uses an SP-100 derived reactor with thermoelectric conversion to supply a continuous 100 kWe. To support full-scale resource utilization, which has power requirements in the megawatt range, a dynamic conversion system is coupled to an equivalent reactor to satisfy power needs for the consolidation phase and beyond.

Assembly and Construction. The characterization of the martian outpost as a permanent facility implies the need for assembly and construction capabilities for long-term structures. Such capabilities include lifting massive payloads, carrying payloads to the emplacement site, and regolith moving/excavating for site preparation. Selected assembly systems and construction equipment also share common functions with machinery for resource mining and lander payload unloading.

The analysis performed for this year's case studies in the area of assembly and construction did not address specific equipment design. Instead, the focus was placed on defining the requirements for outpost construction. Therefore, the equipment selected to support construction operations was estimated from previous studies while other design concepts are still under study. The equipment manifested in the first cargo flight (2005) included a crane to manage off-loading payloads, a digger for surface preparation, and a truck for payload surface transport. Each of these pieces weighs approximately 4 t and supports the construction and assembly during the emplacement and consolidation phases. Under the heading of lander facility upgrades, enhanced off-loading equipment is delivered in 2011 to assist construction and placement of the nuclear plant and oxygen production facilities.

Surface Transportation. Several types of surface transportation are required to perform various functions, including: exploration, local crew transport, local pressurized transport, and long-duration excursions. All rover-type vehicles are powered by fuel cells with reactant regeneration at the outpost. Exploration in the vicinity of the outpost is accomplished by two teleoperated rovers; the first is delivered with the initial crew, and the second is delivered on the third flight with the second crew.

Crew transport within 10 km of the outpost is accomplished with a four-wheeled unpressurized Apollo-type rover and a two-wheeled detachable trailer. The vehicle can be configured to carry two crewmembers and 250 kg of payload or four crew and minimal payload. Two of these rovers are delivered on the first flight. A four-wheeled, pressurized rover provides a crew of four with local pressurized transportation needed to support outpost construction and maintenance. This vehicle carries tools, parts, and EVA suits, and is equipped with an airlock and manipulator arms for IVA access to outpost exteriors. It can also dock directly to habitats thus allowing IVA transfer of crew between rover and habitat. Two of these vehicles are delivered on flights 3 and 4.

In Situ Resource Utilization. It has long been realized that potentially substantial savings/enhancements can be obtained through the use of locally produced items, provided they are of sufficiently high value to justify the overhead of the production facilities. Oxygen exists on Mars (for example) in the form of atmospheric CO_2, and in CO_2 and water ice. It is expected that extraction of this resource will be beneficial through its use as lander fuel and life support makeup. Several concepts for processes exist for martian oxygen production. Atmospheric carbon dioxide electrolysis is a straightforward, site-independent process. As this process has been subject to the most analysis in recent years, it is the best choice as a departure point for further analysis of a total oxygen production outpost architecture.

Scientific evidence indicates that water can be extracted from materials from the surface of Phobos and processed into liquid hydrogen and oxygen. The technology for extracting and processing these resources and the way to best use the propellants are ill-defined and require further study. The selection of a Phobos propellant gateway for this case study was an appropriate departure point for future analysis.

Launch and Landing Systems. Because all Mars landers in this case study are expendable with storable propellants, minimal lander support is required. All launch/landing support equipment is delivered on Flight 4. This consists of off-loading equipment and a set of vehicle servicing equipment. A pressurized tunnel ramp provides IVA crew transfer between the lander and the outpost.

User Accommodations. The function of the user accommodations area is to ensure that user elements receive the required support from the established outpost systems. Support for user activities comes in the form of pressurized volume, power, communications, surface transportation, and crew.

Telecommunications, Navigation, and Information Management. The system initially consists of a single rack of communications equipment in the crew module. Pad markers and navigation beacons are used to mark the landing areas. A ground station will be provided within the constructible habitat.

Figure 3.2.5-1 shows the layout of the Mars outpost. From this figure, relative locations of the power, surface science, launch/landing, in situ resource utilization, and habitat systems can be seen. Figure 3.2.5-2 shows the habitat area and power system in more detail.

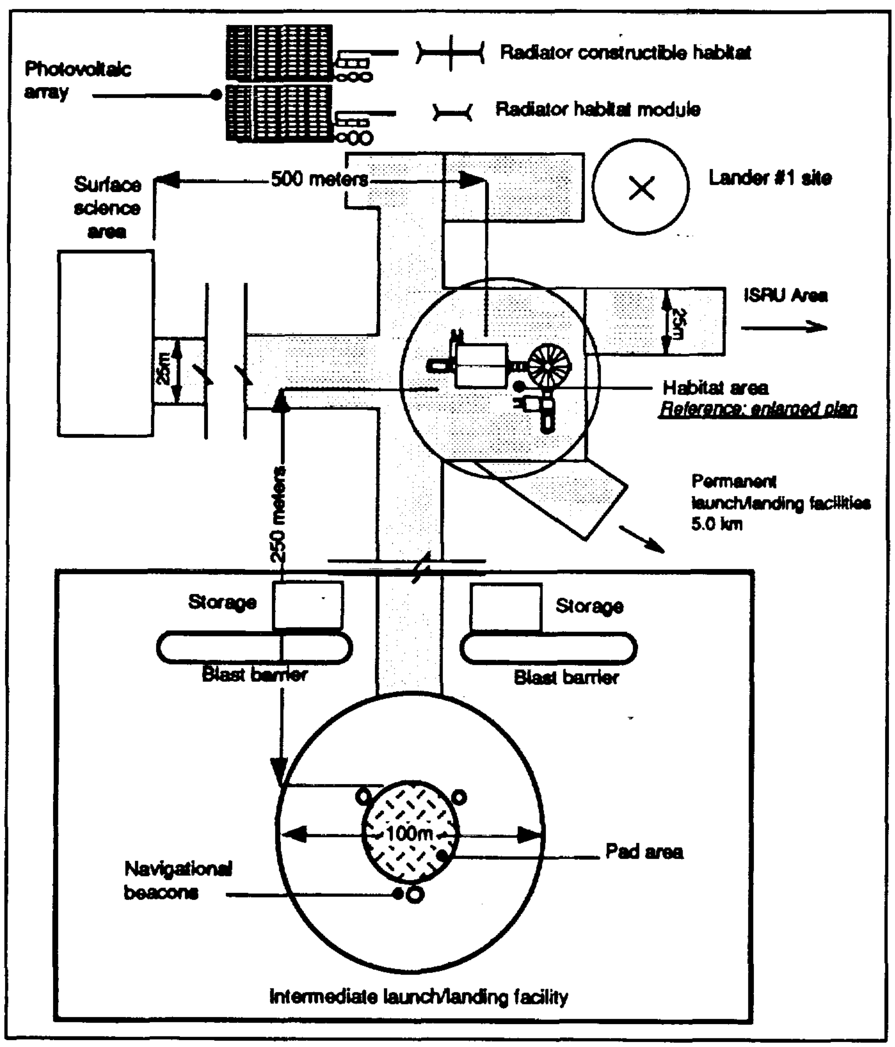


Figure 3.2.5-1.- Mars outpost layout.

3.2.5.2 Enabling Technology

The critical technology development needs for surface systems for the Mars Evolution case study include:

a. Construction technologies

b. Mobile power

c. Surface power (< 1 MWe)

d. Dust contamination control

e. EVA systems technology

f. Phobos water extraction

Construction techniques and system technologies must be developed to support deployment and operations of the Mars Evolution case study. Mobile power systems with power levels of 5 to 40 kWe will be required to support such applications as pressurized long-range manned rovers, construction equipment, and mining/materials transport vehicles. Stationary power technology for the Mars Evolution case study must be provided in support of crew surface activities. Requirements specify output of between 25 and 100 kWe, lightweight construction, erectable or deployable design, and power management/distribution capability.

A method of easily and effectively controlling and removing dust and dirt from EVA suits, airlock seals, and other systems (e.g., solar arrays) is required. A key concern is the introduction of dust into habitable volumes. To meet the EVA requirements of the Mars Evolution case study, development of martian EVA suits and portable life support systems is required. The primary technology needs in support of EVA systems are in the areas of: (1) a low-mass, portable life support system, and (2) materials development for suit, glove, and visor that are resistant to abrasions by dust over long periods of time and that also facilitate cleaning methods.

The technology for extracting water from the surface of Phobos is currently undefined, but will be complex and require significant research and development activities. Technology development will be required in the areas of mining, beneficiation, processing, collection, purification, and storage.

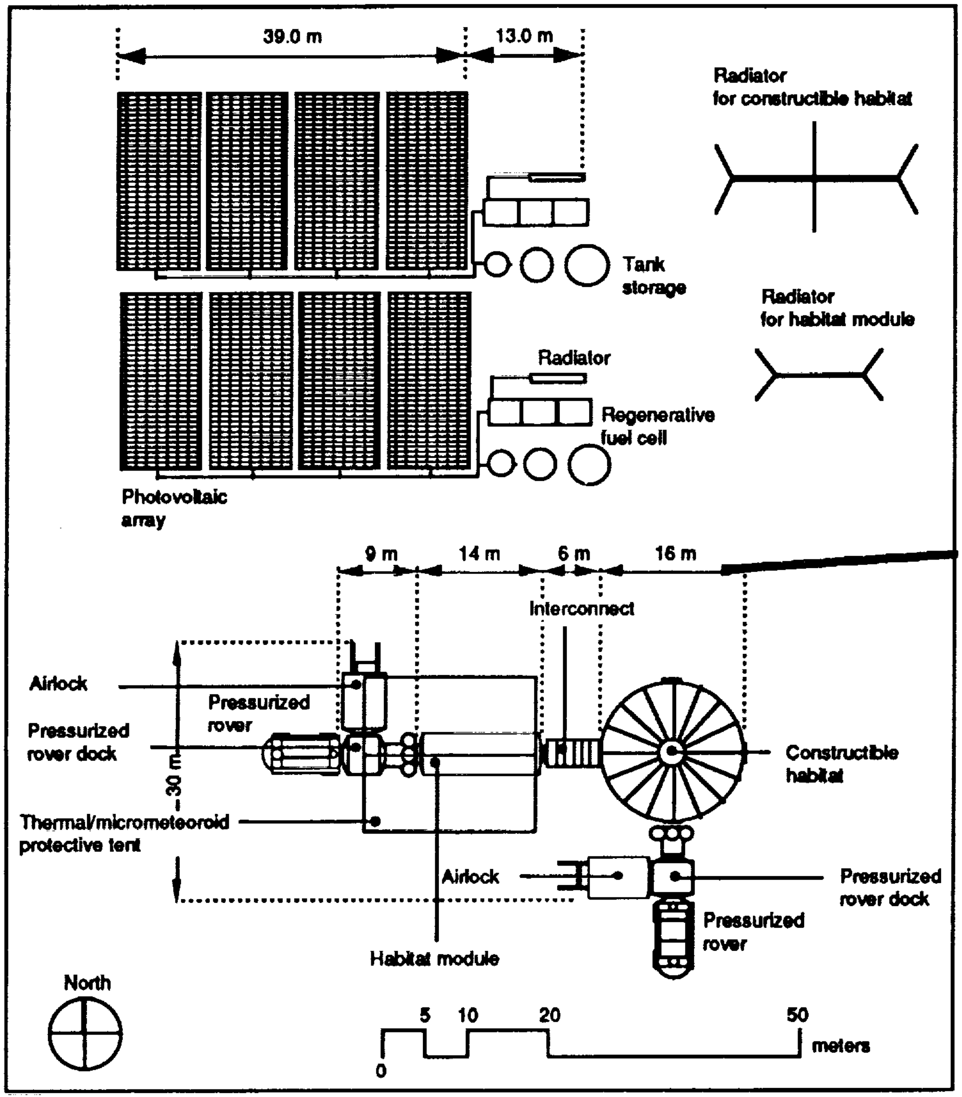


Figure 3.2.5-2.- Mars outpost habitat and power system layout.

3.2.5.3 System Alternatives

Many studies were carried out to select the surface systems used in this case study. In Volume III of this annual report, Section 5 (Distributed Studies of Equipment and Subsystems) discusses in detail the designs and characteristics of planetary surface systems.

3.2.6 Synthesized Mission Manifest

The buildup of LEO supporting infrastructure begins in the year 2000 with Space Station Freedom growth elements used to conduct Controlled Ecological Life Support System (CELSS) and artificial-g research. The first elements of the separate node assembly fixture, which serves as the Mars transportation depot, arrive at LEO in 2001. The first elements of the cargo vehicle are delivered to LEO in 2002 to support the 2005 launch date. The first elements of the piloted vehicle, which is launched in 2007, arrives at LEO in 2004. A schedule for other major components of the case study, including development, flight test, and system verification, is shown in figure 3.2.6-1.

The mission manifests for the Mars Evolution case study (flights 1 through 8) are shown in figures 3.2.6-2 through 3.2.6-9. These figures include manifests for ETO flights, the Mars transfer vehicle in LEO prior to trans-Mars injection (TMI), and the payload elements delivered to Mars and/or Phobos.

The Mars Evolution case study assumes that ETO transportation is accomplished by Shuttle flights delivering crew and small payloads, and a heavy lift launch vehicle delivering large payloads and propellant to a Space Station Freedom-compatible orbit. Mars transfer vehicles deliver crew and payloads from LEO to Mars orbit, and Mars excursion vehicles transfer crew and payload to the Mars surface. The mass required in LEO each year to support the scenario is summarized in figure 3.2.6-10. This figure also summarizes the ETO flights required per year. Averaged over the entire scenario lifetime, the total mass to LEO requirement is well within the capacity of four HLLVs. The dry mass to LEO, averaged over the same 13-year period, is 122 t/year.

The upper portion of figure 3.2.6-11 shows the initial mass in LEO requirements for each flight to Mars, and the lower portion shows how each flight's mass was manifested over time to arrive at the results presented in figure 3.2.6-10. Each shaded box on the lower half of figure 3.2.6-11 represents the masses associated with one flight to Mars. The average mass per flight is about 750 t, and ranges from 552 t in 2005 to 1,052 t in 2007. The mass to LEO requirements for the Mars Evolution case study provide understanding of the amount of on-orbit operations that will be required for vehicle assembly.

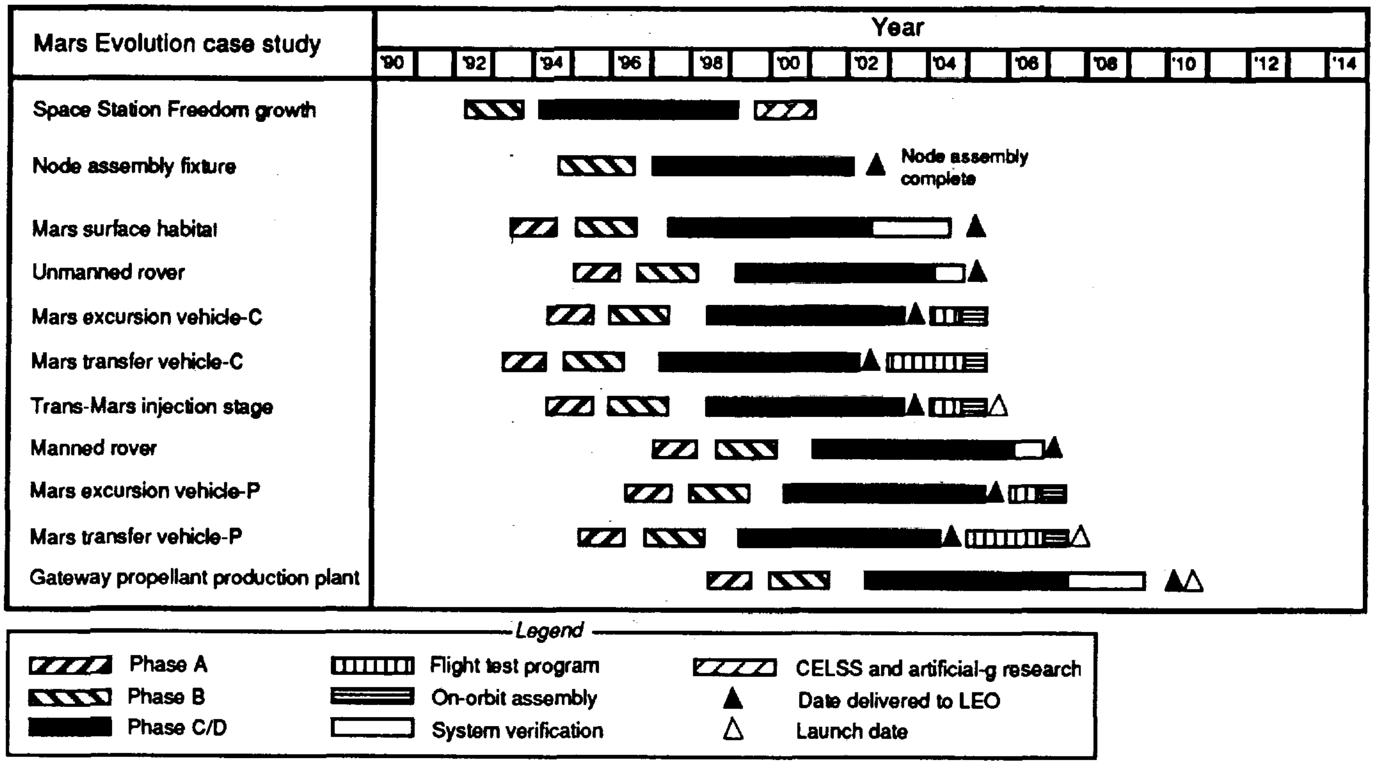


Figure 3.2.6-1.- Mars Evolution case study schedule.

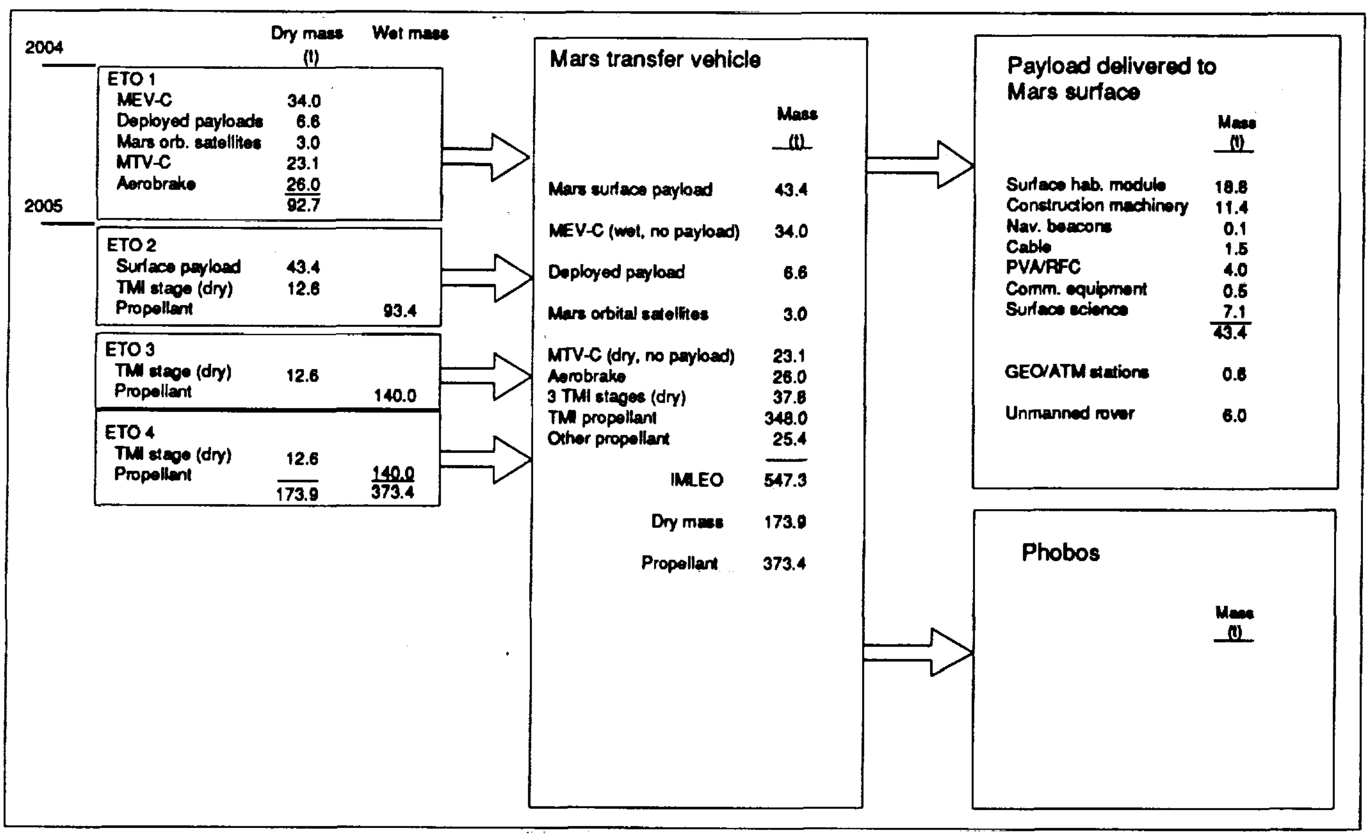


Figure 3.2.6-2.- Mars Evolution manifest: Flight #1—cargo flight, 2005.

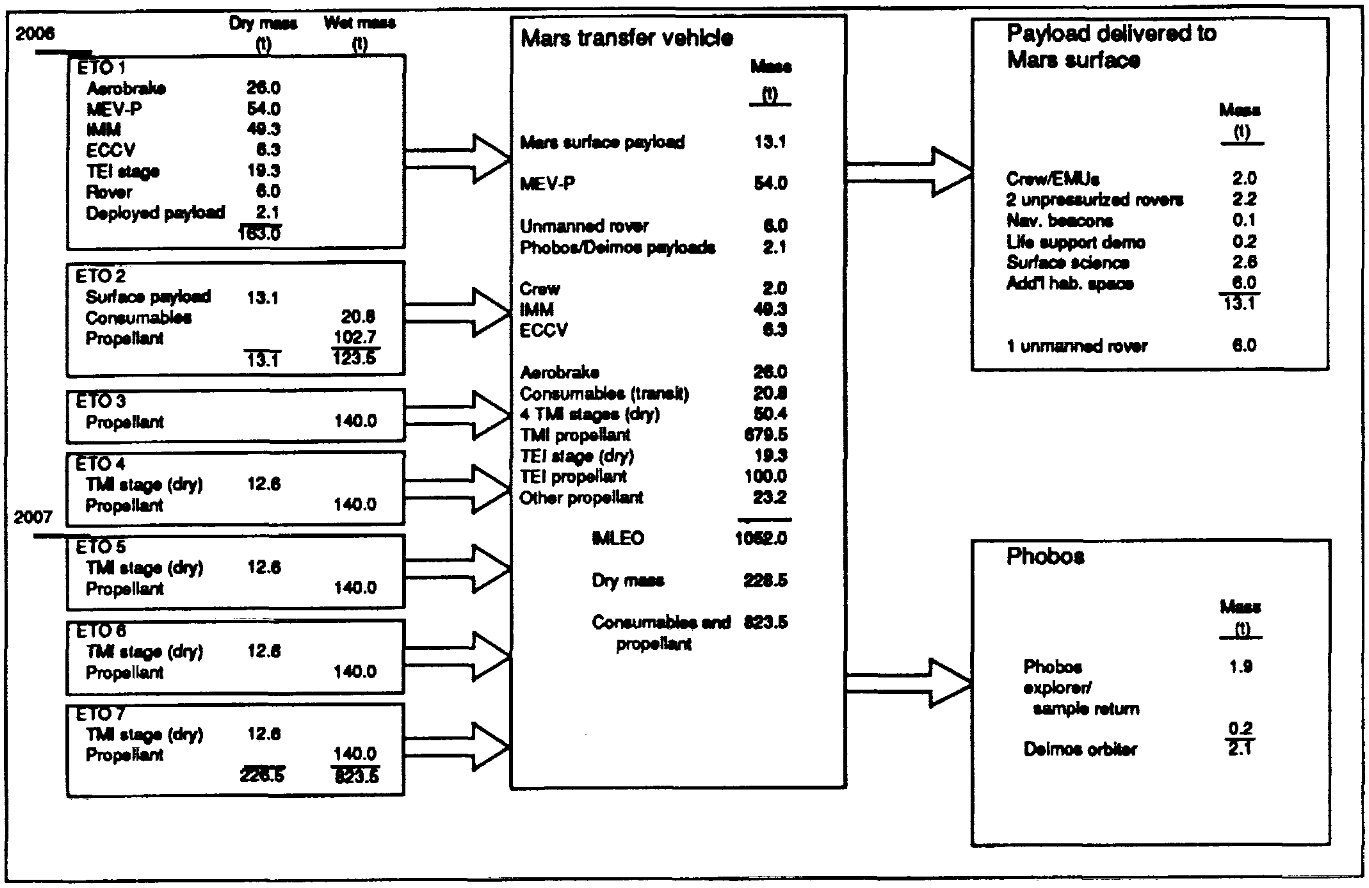


Figure 3.2.6-3.- Mars Evolution manifest: Flight #2—piloted flight, 2007.

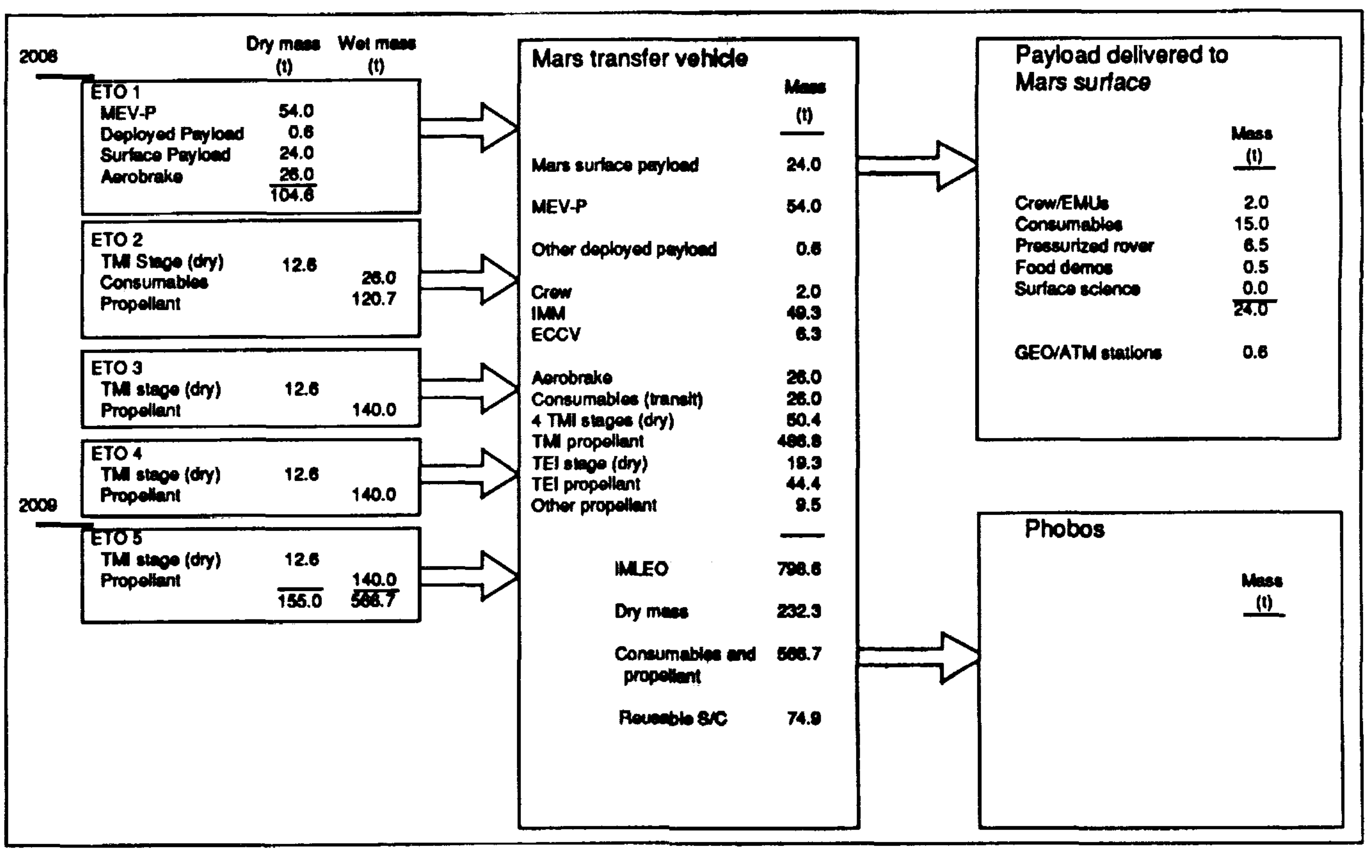


Figure 3.2.6-4.- Mars Evolution manifest: Flight #3—piloted flight, 2009.

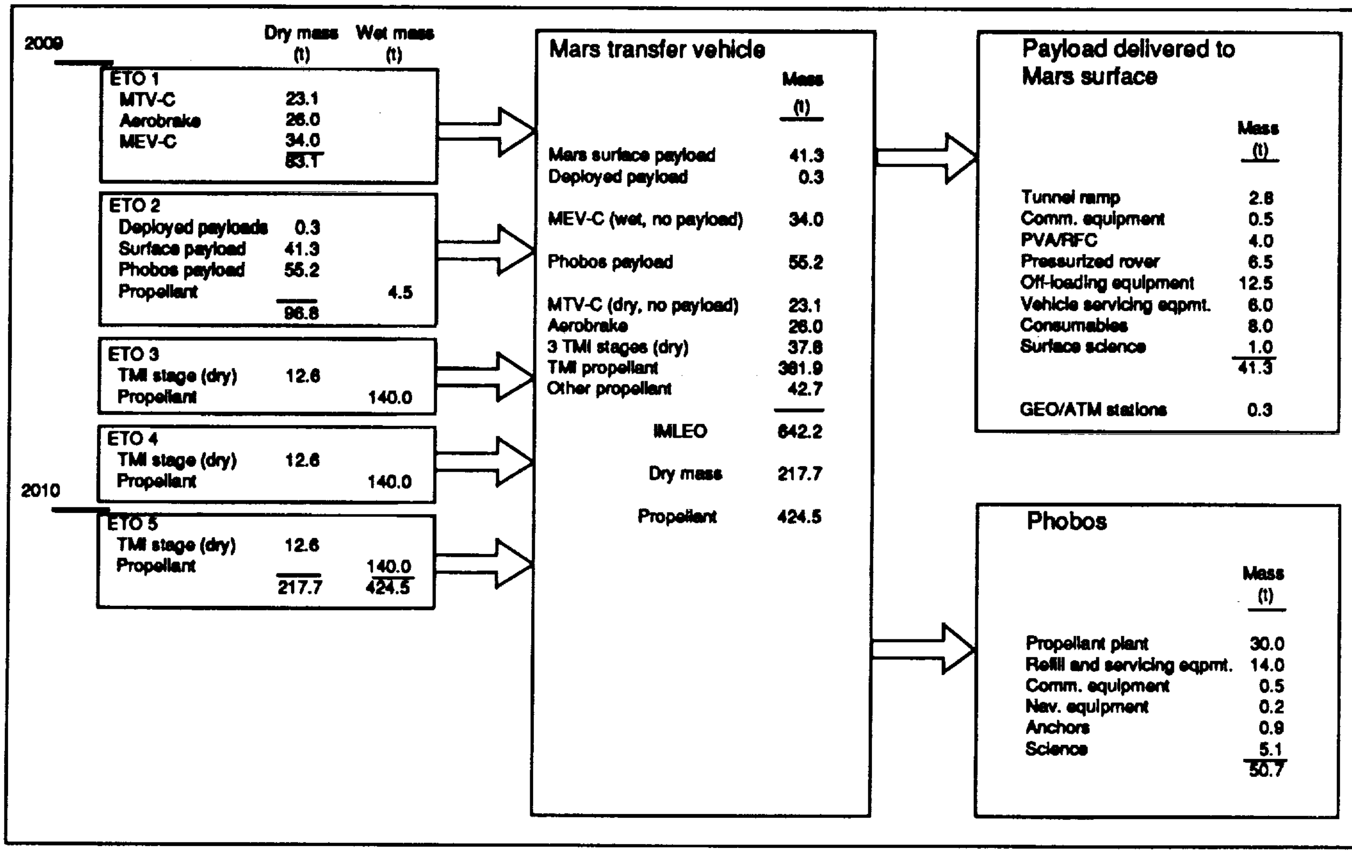


Figure 3.2.6-5.- Mars Evolution manifest: Flight #4—cargo flight, 2010.

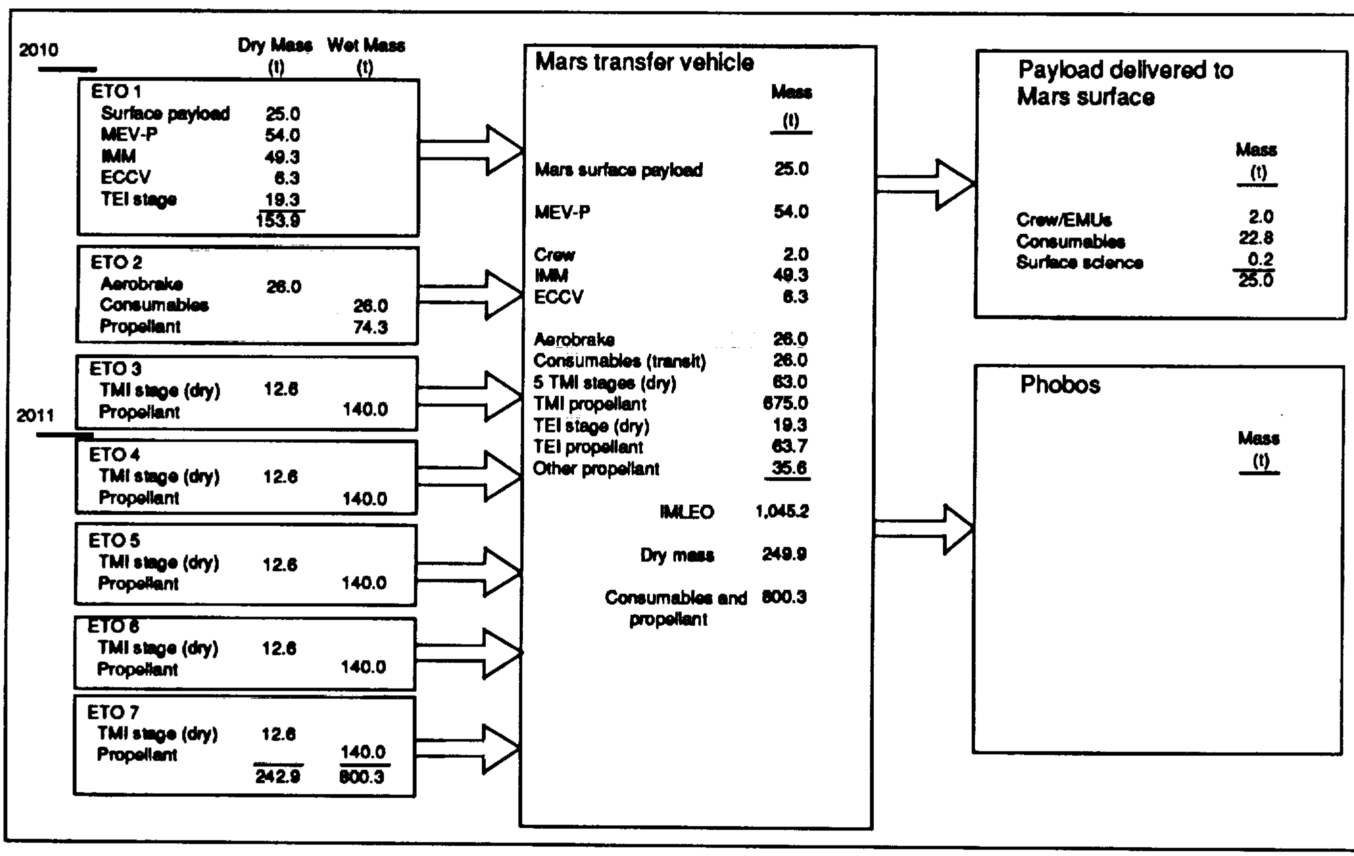


Figure 3.2.6-6.- Mars Evolution manifest: Flight #5—cargo flight, 2011.

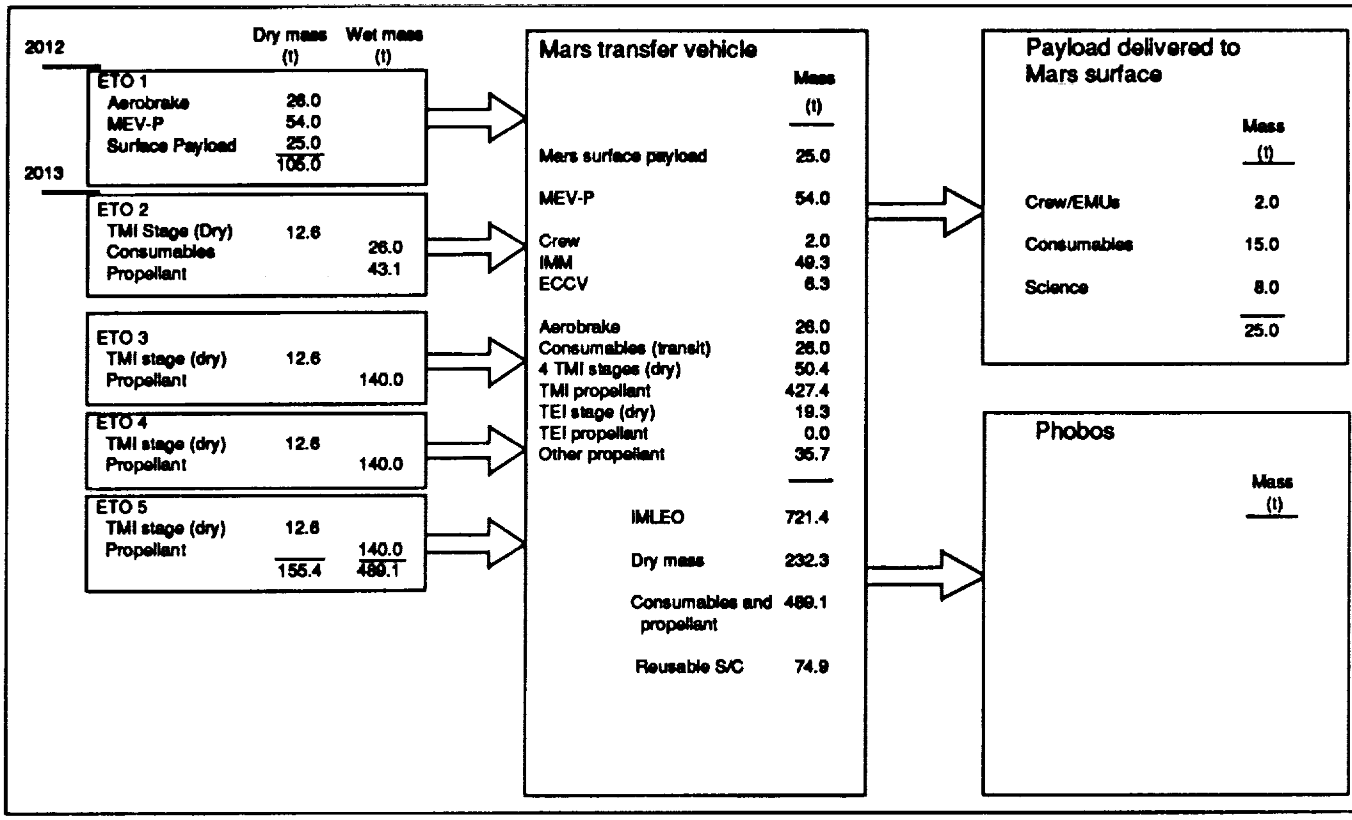


Figure 3.2.6-7.- Mars Evolution manifest: Flight #6—piloted flight, 2014.

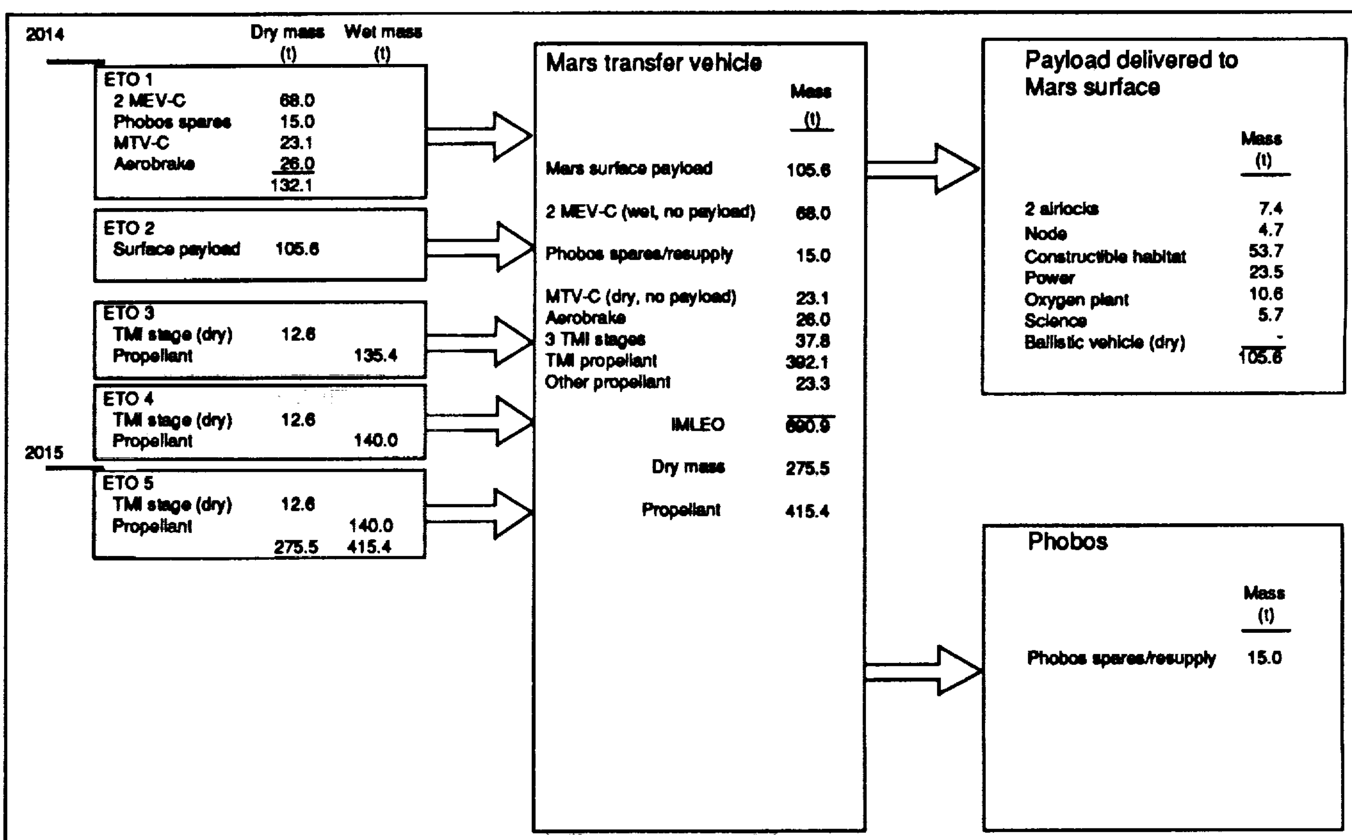


Figure 3.2.6-8.- Mars Evolution manifest: Flight #7—cargo flight, 2016.

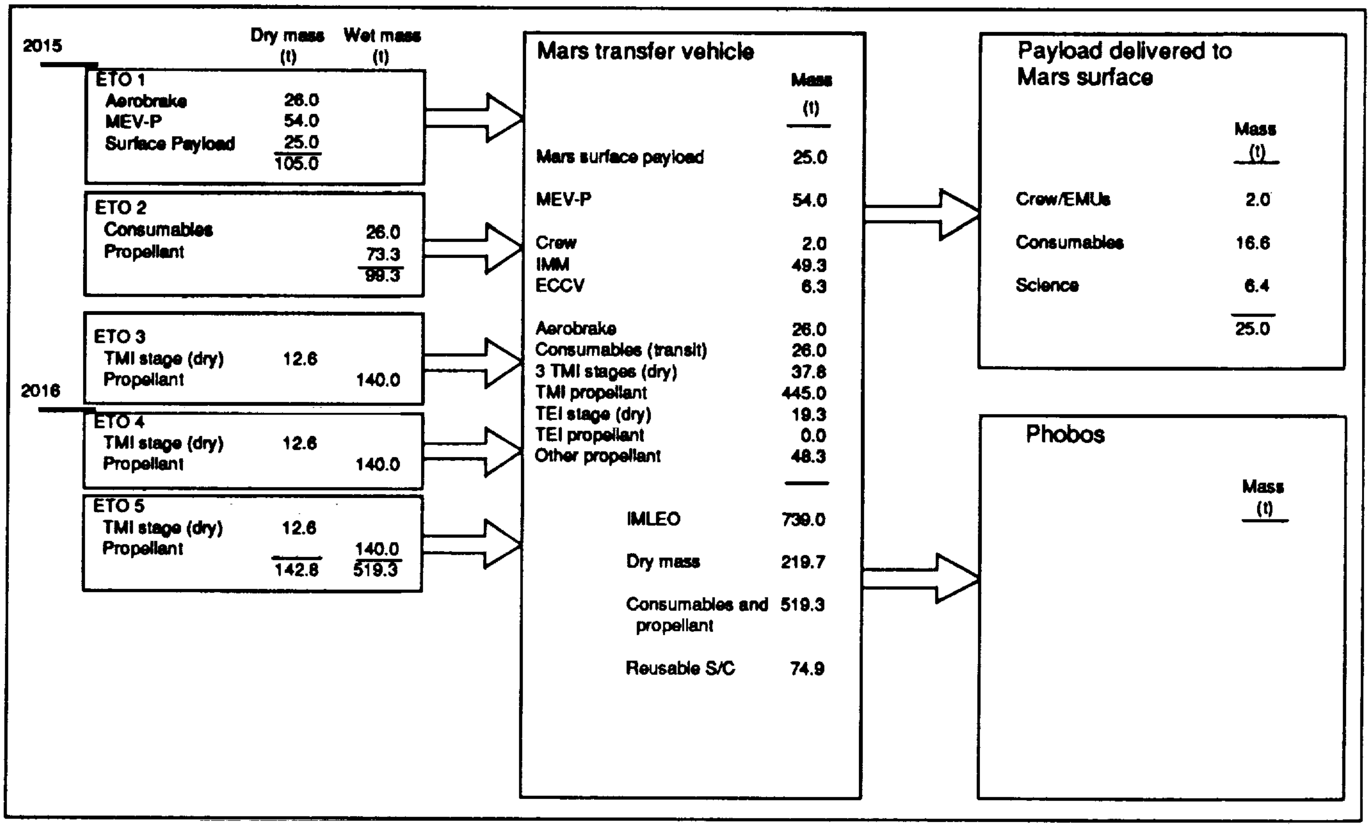


Figure 3.2.6-9.- Mars Evolution manifest: Flight #8—piloted flight, 2016.

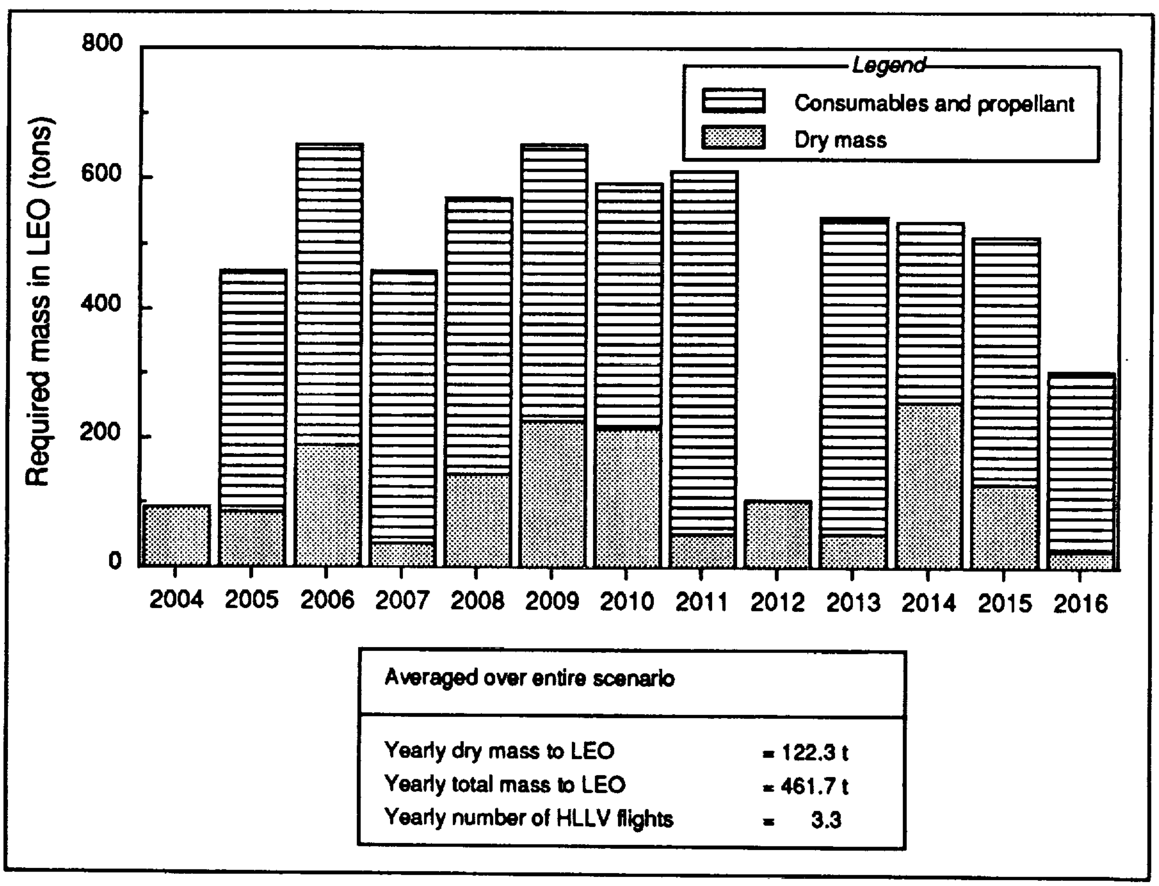


Figure 3.2.6-10.- Mars Evolution annual mass in LEO requirements.

Launch date	8/05	9/07	10/09	11/10	12/11	1/14	2/16	3/16	
Flight	1	2	3	4	5	6	7	8	Total
Dry mass (t)	173.9	226.5	155.0	217.7	242.9	155.4	275.5	142.8	1,589.7
Wet mass (t)	373.4	823.5	566.7	424.5	800.3	489.1	415.4	519.3	4,412.2
Total	547.3	1,050.0	721.7	642.2	1,043.2	644.5	690.9	662.1	6,001.9
No. of ETO	4	7	5	5	7	5	5	5	43

	2004	2005	2006	2007	2008	2009	2010	2011	2012	2013	2014	2015	2016	Total	Avg.
Dry mass (t)	92.7	81.2	188.7	37.8	142.4	12.6 205.1	12.6 192.5	50.4	105	50.4	262.9	12.6 117.8	25.2	1,589.7	122.3
Wet mass (t)	0	373.4	409.5	420.0	426.7	140.0 284.5	140.0 240.3	560.0	0	489.1	275.4	140.0 239.3	280.0	4,412.2	339.4
Total mass (t)	92.7	454.6	592.2	457.8	569.1	642.2	585.4	610.4	105	539.5	538.3	509.5	305.2	6,001.9	461.7
No. of ETO	1	3	4	3	4	5	4	4	1	4	4	4	2	43	3.3

Figure 3.2.6-11.- Mars Evolution manifest: mass.

3.3 MARS EXPEDITION CASE STUDY

The primary objective of this case study is to embark upon the first human expedition to the surface of Mars. The case study assumes a single expedition to Mars, undertaken at the earliest possible opportunity. In addition, the environment, geological features, and material of Mars would be studied to advance knowledge of the origin of the solar system and survey the potential for using martian resources.

3.3.1 Key Features

Key features of this case study include:

a. An expendable vehicle strategy in which a single vehicle, with all supporting systems and landing/ascent vehicle attached, is launched intact into LEO. Several launches then bring up the trans-Mars and trans-Earth propulsion stages.

b. A three-member crew is launched to Mars in a zero-gravity vehicle on an opposition-class trajectory with a free flyby abort capability.

c. The vehicle uses aerobraking at Mars and remains in Mars orbit for 30 days.

d. A Mars lander, with all three crewmembers, descends to the surface for 20 days.

e. Limited science equipment is carried to the surface of Mars.

f. After their tour of duty is completed, the crew returns for direct entry to Earth's surface.

g. The nominal mission duration is less than 18 months.

3.3.1.1 Mission Profile

Two approaches were considered for this case study: (1) "splint/sprint" and (2) "all-up." In the split/sprint approach, one Mars transfer vehicle carries the crew and some hardware, and a second vehicle carries the remainder of the hardware. For the all-up approach, all the expedition hardware and crew are dispatched on one Mars transfer vehicle. Trade studies showed that the all-up approach is the better of the two, when considering total mass to LEO and mission safety/success; therefore, this approach was selected as the baseline. The strategy presented here consists of a single mission to Mars with 11 phases, as described below and illustrated in figure 3.3.1-1.

1. **Earth to orbit:** In this phase, one completely assembled Mars expedition vehicle is launched into LEO. Propulsion stages for the trans-Mars and trans-Earth injection burns are launched separately, and they dock with the expedition vehicle. It is assumed that heavy lift launch vehicles with a payload mass capability of at least 140 t delivered to LEO are used. The cylindrical payload shroud dimensions on the launch vehicle are 12.5 meters in diameter by 25 meters in length. The Earth launch places the Mars vehicles into a circular 500 km altitude orbit at 28.5 degrees inclination. Other orbital parameters are selected to give the largest possible departure window.

2. **Earth orbit operations:** Operations in Earth orbit include any rendezvous maneuvers required for crew transfer and stage docking following the Mars vehicle launch. The case study ground rules assumed that no on-orbit assembly would be required. Also included are any orbit adjust maneuvers required to modify the orbital parameters for the LEO departure maneuver.

3. **Earth to Mars transfer:** The transfer from Earth to Mars begins with the TMI maneuver, which places

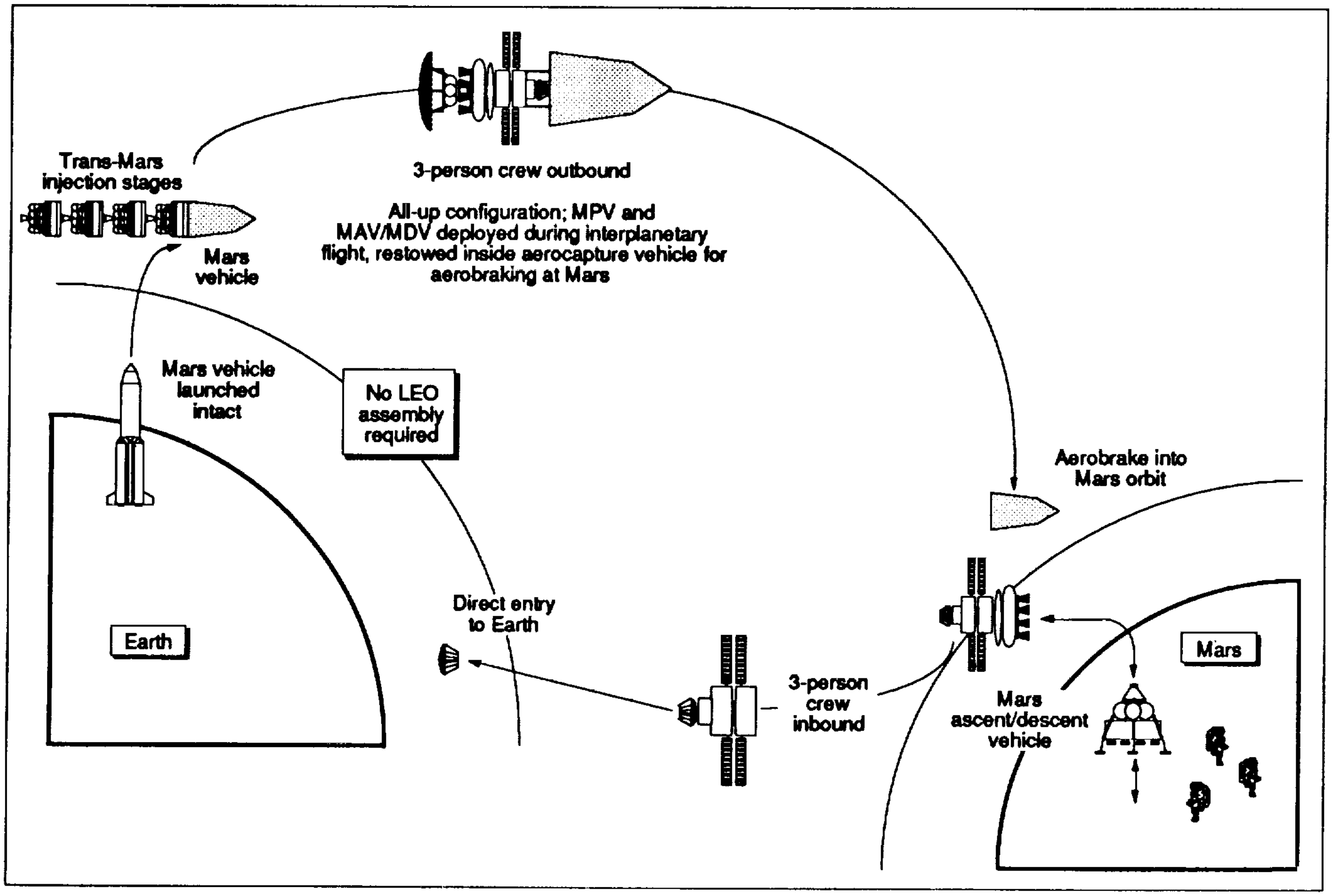


Figure 3.3.1-1.- Mars Expedition mission profile.

the vehicle on a trajectory that will either fly directly to Mars or, if desirable for a particular opportunity, fly to Mars using a Venus swingby. A free Mars flyby abort capability must exist for the manned vehicle. After the TMI burn, the TMI stage (engines, tanks, and associated systems) is discarded. Any midcourse corrections must be performed by the reaction control system.

4. **Mars capture:** Upon arrival at Mars, the vehicle performs an aerobraking maneuver to reduce the trajectory energy and enter an elliptical orbit around Mars. Upon reaching apoapsis, the periapsis is raised to an altitude of 500 km to avoid reentering the martian atmosphere. A plane change maneuver is done while in the atmosphere on the first pass to place the vehicle into an orbit compatible with landing and trans-Earth injection (TEI) departure requirements. Immediately after the aerobraking maneuvers are completed, the aerobraking shield is jettisoned.

5. **Prelanding Mars orbit operations:** Preparations for landing are made after the vehicle is in Mars orbit. Five days are allotted for these preparations, which include detaching the Mars descent vehicle from the aerobrake shell, deploying any orbiting satellites, and preparing the Mars descent/ascent vehicles.

6. **Descent and landing:** Descent to the surface of Mars begins with a maneuver that lowers periapsis into the atmosphere. Aerobraking maneuvers are performed in the atmosphere to reduce entry velocity and arrive at the landing site. After the aerobraking phase, the terminal descent engines are turned on to reduce the vehicle's energy for landing and to perform any hazard avoidance maneuvers required.

7. **Mars surface activities:** During the 20-day stay on the martian surface, the crew reconnoiters the local environment. A geophysical station is deployed to measure properties such as seismicity, magnetic fields, gravity fields, heat flow, atmospheric composition, and meteorology. A total of five EVAs (four with rover traverses) explore the surface and subsurface geology for both scientific and engineering purposes. Human biomedical adaptations to the martian environment are also studied.

8. **Ascent and docking with orbiter:** After the 20-day stay on the surface, the three crewmembers, equipment, and surface samples ascend from the martian surface and rendezvous with the interplanetary vehicle.

9. **Mars orbit pre-departure operations:** Five days are allotted to prepare the interplanetary vehicle for TEI. This preparation includes disposal of the ascent vehicle and any final orbit shaping prior to the departure maneuver.

10. **Mars to Earth transfer:** The interplanetary vehicle performs the TEI maneuver and departs on a return trajectory to Earth. This departure is either a direct return trajectory or a Venus swingby. The TEI stage is discarded after the TEI burn. Any midcourse corrections are made with the reaction control system.

11. **Earth capture, entry, and landing:** As the interplanetary vehicle approaches Earth, the crew and samples transfer into the Earth entry vehicle, and the interplanetary vehicle is discarded. The entry vehicle enters Earth's atmosphere and performs aeromaneuvers to effect a direct entry to landing.

Four acceptable mission opportunities, shown in table 3.3.1-I, occur between the years 2000 and 2010. Based on programmatic considerations, the 2004 opportunity was chosen as the reference. All outbound trajectories have a free Mars flyby abort (free-abort) capability (in the event of a failure involving the TEI stage) and a powered abort option (in the event of some other failure). A Venus swingby is used on either the outbound or return trajectory for the mission.

An important trade was the comparison of the split/sprint strategy to the all-up approach. In addition, two alternative strategies were considered: one strategy with the vehicles going to and from a 500-km circular martian orbit and the other with the vehicles in a 250 km by 33,809 km ellipse (1 sol). Table 3.3.1.-II shows a comparison of the results of this trade study. As shown in table 3.3.1-I, there is a subset of trajectories within the general class of 500-day missions that permit free return and have energetic outbound requirements comparable to TMI requirements for the 1,000-day class missions. Thus, the need to split missions between crew and cargo with the added operational complexity is removed. Also, the TEI stage is removed from the cargo vehicle for mission safety considerations. Removing the TEI stage from the cargo vehicle (which differs from the FY 1988 split/sprint concept) also enforced the decision to go to the all-up configuration. The split/sprint approach requires a total of seven ETO cargo flights and one Space Shuttle personnel flight. The all-up missions require six cargo flights and one Shuttle flight. Although the 250 km x 1 sol orbit requires slightly less mass, the Mars orbital operations are more complex for the highly eccentric orbit. Therefore, the all-up, 500-km circular Mars orbit mission was chosen.

TABLE 3.3.1-I.- MARS EXPEDITION REFERENCE TRAJECTORY OPTIONS

		Reference		
Launch date	2 Sep 2002	10 Jun 2004	26 Sep 2007	25 Nov 2010
Departure declination	-21.3 deg	-5.9 deg	37.0 deg	3.9 deg
Mars arrival date	15 Jun 2003	11 Apr 2005	25 Feb 2008	17 Sep 2011
Mars departure date	15 Jul 2003	11 May 2005	26 Mar 2008	17 Oct 2011
Nominal mission duration	485 days	520 days	510 days	520 days
Free abort duration	600 days	671.1 days	727.8 days	696.3 days
Powered abort duration	380 days	470 days	655days	490 days
ΔV trans Mars injection	4.195 km/s	4.140 km/s	4.350 km/s	4.368 km/s
ΔV (500 km) trans Earth injection	3.836 km/s	3.839 km/s	3.614 km/s	2.793 km/s
ΔV (1 sol) trans Earth injection	3.109 km/s	2.722 km/s	3.470 km/s	1.984 km/s
ΔV powered abort	3.038 km/s	0.969 km/s	2.335 km/s	1.739 km/s
Nominal outbound duration	285.7 days	304.6 days	152.6 days	295.9 days
Nominal return duration	169.3 days	185.4 days	327.4 days	194.1 days
Venus swingby	Outbound	Outbound	Return	Outbound
Nominal Earth entry velocity	12.104 km/s	13.253 km/s	12.722 km/s	13.303 km/s

TABLE 3.3.1-II.- COMPARISON OF MANIFESTING OPTIONS

	Split/Sprint	"all up" (500 km altitude circular orbit at Mars)	"all up" (1 sol period elliptical orbit at Mars)
Cargo	209 t	0	0
Piloted	616 t	776 t	762 t
Total	825 t	776 t	762 t

* Values above are mass in low-Earth orbit

The mission uses expendable vehicles with minimal infrastructure that is not mission specific. The Space Shuttle is used as Earth-to-orbit transportation for the flight crews, and heavy-lift launchers are used for all other ETO transportation. Transportation requirements occur over a short period, with a total ETO mass requirement of 776 t.

Major milestones for the Mars Expedition case study are shown in figure 3.3.1-2. The following programmatic template was assumed: 1-year Phase A, 6-month procurement, 18-month Phase B, 1-year procurement, 5-year Phase C/D, 1 to 2 years for flight test, on-orbit mating, and system verification. This template, coupled with the FY 1992 budget as the earliest fiscal cycle that can be modified, indicates that the earliest feasible launch opportunity is 2004.

3.3.2 Science Opportunities and Strategy

3.3.2.1 Opportunities

The Mars Expedition offers a variety of unique scientific opportunities to study the physical properties and processes of Mars, including its fundamental nature, history, geomorphological development, mode of origin, resource potential, and natural environmental characteristics. The science opportunities for the Mars Expedition, while similar in theme to those of Mars Evolution, are limited in scope due to a surface stay time of 20 days.

Potential investigations span the science disciplines and include geological and geophysical studies, the study of rock and sediment composition, distribution, age (analysis done in Earth-based laboratories), the study of subsurface properties, hydrology, erosion phenomena, dust mobility, cosmic ray flux, and regolith maturity.

In addition, the possibility of past or extant life on Mars prompts exobiological studies, including the search for exposed sedimentary deposits and associated fossils, organic compounds, and biogenic elements. Relict lake beds are also desirable areas to look for fossil life-forms.

A surface geophysical monitoring station measures atmospheric volatiles, composition, structure, spatial and diurnal variations, and long-term temporal variations. The monitoring stations also collect data on winds, clouds, dust storms, and dust density and composition. Other studies made by a surface station include cosmic rays, magnetometry, heat flow, and passive seismology.

The Mars lander provides the only pressurized laboratory during the Mars Expedition. For this reason, anything more than rudimentary geochemical/exobiological analyses will be performed on Earth. Remote sensing from the rover using an electromagnetic sounder will provide some information on local resources in support of resource evaluation of martian materials.

Life sciences experiments on the surface include the study of human performance, behavior, and physiology in the Mars environment, and the biological effects of

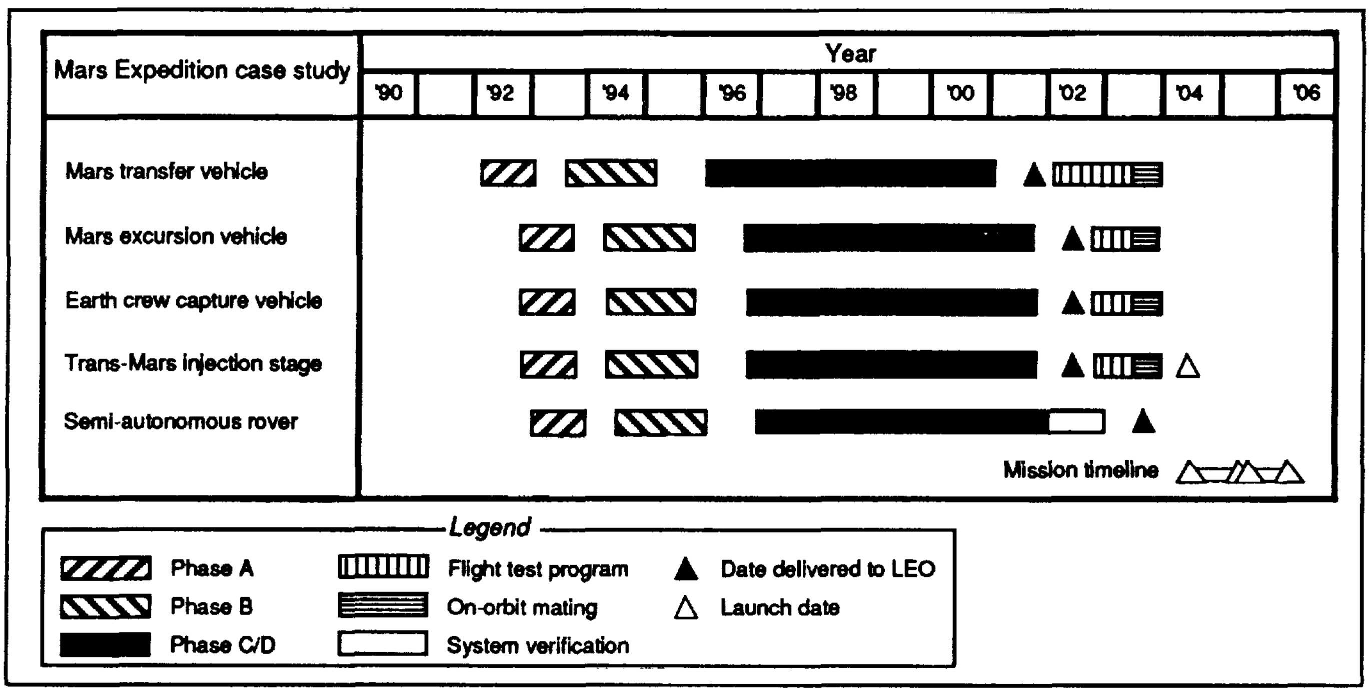


Figure 3.3.1-2.- Mars Expedition development and mission schedule.

radiation and dust from a short-term Mars stay, using limited instrumentation within the Mars lander.

Since the travel time between Earth and Mars is long, significant science investigations, referred to here as "cruise science," would be done en route. In particular, studies of human response to radiation and the microgravity environment and observations of human physiology/psychology would be conducted. In addition, a full suite of particles and fields experiments would be possible, including observations of the solar wind, the Sun, cosmic rays, and gamma-ray and X-ray bursts. Astronomical observations could be conducted at optical, IR, and UV wavelengths.

A more detailed treatment of science opportunities is included in Volume VI of the FY 1989 Exploration Studies Technical Report.

3.3.2.2 Strategy

The Mars Expedition case study allows 30 days in the Mars system, 20 days on the Mars surface, and three crewmembers to the surface. The basic strategy for accomplishing the science objectives is described below.

While on the surface, the crew (1) conducts geologic and geophysical observations near the landing site on foot and by rover, collecting rock, sediment, and exobiology samples; (2) deploys instrumentation for short-duration experiments that will be completed before the crew leaves the surface (seismic tests, atmospheric balloons and rockets, microbe/bacterial/plant organics exposure tests); and (3) deploys instrumentation to be left on Mars for later study and analysis to measure martian properties and processes that can be monitored from Earth on a long-term basis; i.e., a geophysical/atmospheric sciences station.

The science payload includes:

a. Instrumented manned unpressurized rover

b. Visible/IR imager (close-range telescope and camera)

c. Geologic exploration equipment (hammers, shovels, scoops, rakes, sieves, tongs, optical and CCD camera, shallow coring device, drive tubes, trenching instruments, penetrometer, sample bags, etc.)

d. Portable geophysics traverse package (shallow seismic experiments, traverse magnetometer, gamma ray, neutron, visible and near IR spectrometer, mass spectrometer, electromagnetic sounder.)

e. Geophysical/atmospheric station

f. Meteorological balloons (4)

g. Biomedical laboratory instruments (health maintenance, observations, experimental testing)

3.3.2.3 Surface Science Scenario

This discussion illustrates how the science objectives could be reasonably executed on the martian surface. The description provides some operational scope and reality to the science mission in terms of objectives, stay time, EVA limits, crew time, and surface vehicle availability.

The landing site is a starting point from which the operations can commence, in order to scope the activity. Although the chosen site is a valid candidate, it was selected as typical and does not imply advocacy.

The example Mars Expedition landing site is near the equator at 8.3 degrees south latitude and 44.2 degrees west longitude in an area known as the Ganges Chasma. Spectacular landslides, some enormous, occur along most of the chasma walls. The landing site area provides an opportunity to sample material from the landslides from each of the north and south chasma walls, thereby providing valuable stratigraphic cross-sectional information regarding the geological structure and history of Mars formation.

The first 6 days after landing, the crew remains in the lander to acclimate from zero-gravity trans-Mars flight to the Mars surface gravity, which is approximately one-third that of Earth. During this 6-day acclimation period, the crew will perform a series of very important tasks:

a. Immediately upon landing, they deploy a rover that the crew operates remotely. The rover supports the inspection of the lander for any structural damage and/or Mars surface stability conditions that may necessitate an abort to orbit and collects a contingency soil and rock sample.

b. The crew photographs the Mars terrain through the lander windows for documentation purposes in the event that an abort is necessary before crew surface activity commences.

c. The crew takes a series of photographs from the open hatch in top of the lander, producing a panorama of the martian landscape under different sunlight conditions.

d. From the open hatch on top of the lander, the crew will take a series of photographs and infrared measurements of the top sections of north and south canyon walls under different lighting conditions.

On the ninth day, the crew exit the lander for their first EVA to check the structural integrity and surface stability of the lander, collect more contingency samples, and

deploy the Mars surface experiment package. The package is composed of a set of geophysical instruments to measure the surface and subsurface properties of Mars and a set of instruments to measure the atmospheric properties and meteorological conditions of Mars.

Four EVAs using the rover are dedicated to performing geological sampling and mapping of the martian surface. The first geological traverse is to the North Chasma Wall Landslide, then to the South Chasma Wall Landslide, then to the east to a boulder and a crater and their respective wind streaks, and finally to the west across a dunes field to a couple of large boulders. A geologic map of the landing site area showing the traverses and their direction, distance, and number of samples collected is shown in figure 3.3.2-1.

During their stay on the surface, the crew also deploys four meteorological balloons. The balloons carry a small lightweight instrument to measure the temperature, pressure, wind direction, and wind velocity of the Mars atmosphere as a function of altitude.

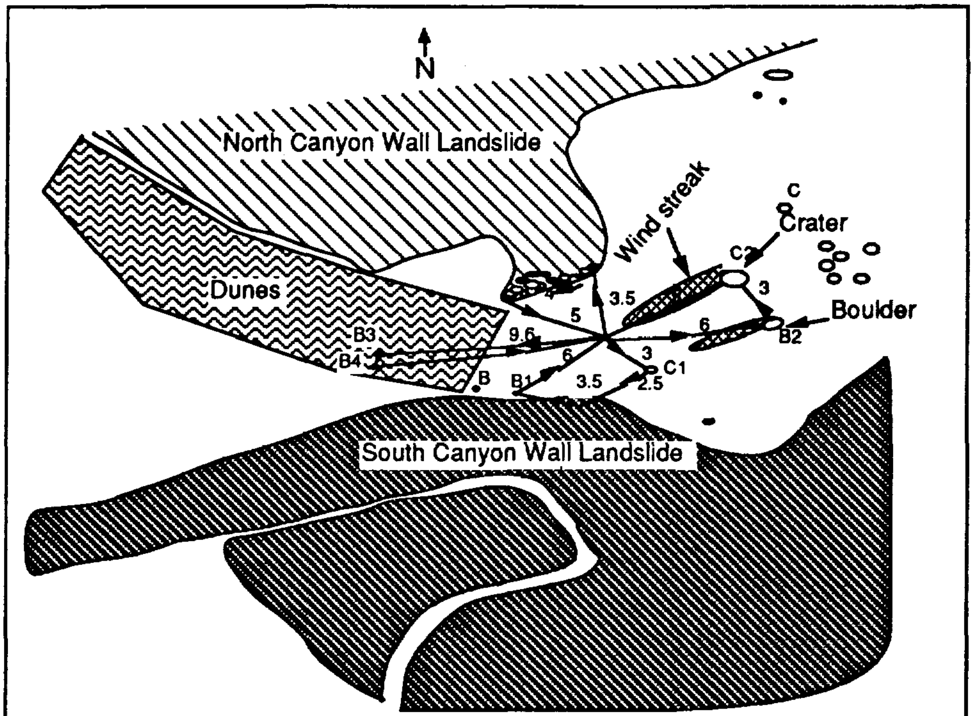
3.3.3 Transportation Systems

Transportation systems for the Mars Expedition case study are designed to achieve a human landing on Mars at the earliest feasible opportunity. The vehicles accommodate a three-member crew transferring to Mars with a free Mars flyby abort capability. The vehicles do not use artificial gravity, and the crew enters Earth's atmosphere directly on the return leg. Aerobraking is utilized for entry into Mars orbit and for landing on the Mars surface. This case study assumes an all-up configuration that uses a single, expendable Mars transfer vehicle, which is launched empty but intact to LEO with an HLLV.

3.3.3.1 Elements and Systems

Transportation elements and systems include ETO systems and space transfer vehicles. For this case study, the Mars transfer vehicle (MTV) is assumed to perform the transfer from Earth orbit to Mars orbit. To deliver the MTV to Earth orbit, six HLLV launches are required. One flight brings up the MTV without propellants, and the other five deliver the trans-Mars and trans-Earth injection stages and propellants. On-orbit activities include propellant transfer to the four trans-Mars injection stages and the trans-Earth injection stage, and connection of these stages to the MTV.

The MTV is designed to carry three crewmembers from Earth orbit to Mars orbit. The vehicle is shown in figure 3.3.3-1. The MTV does not provide artificial gravity and uses LH_2/LOX chemical propulsion. Four trans-Mars injection stage (TMIS) modules are needed to inject the MTV into Mars orbit. Upon entering Mars orbit, the MTV utilizes an aerobrake



North Wall Landslide

Traverse number	Traverse distance (km)	Number of samples
1	3.5	14
2	4.0	16
3	5.0	18

South Wall Landslide

Traverse number	Traverse distance (km)	Number of samples
1	3.0	11
2	2.5	8
3	3.5	11
4	6.0	18

East - Boulder - Crater

Traverse number	Traverse distance (km)	Number of samples
1	6.0	19
2	3.0	10
3	5.0	16

West - Dunes - Boulders

Traverse number	Traverse distance (km)	Number of samples
1	9.6	24
2	9.6	24

Figure 3.3.2-1.- Mars Expedition Ganges Chasma landing site geology exploration.

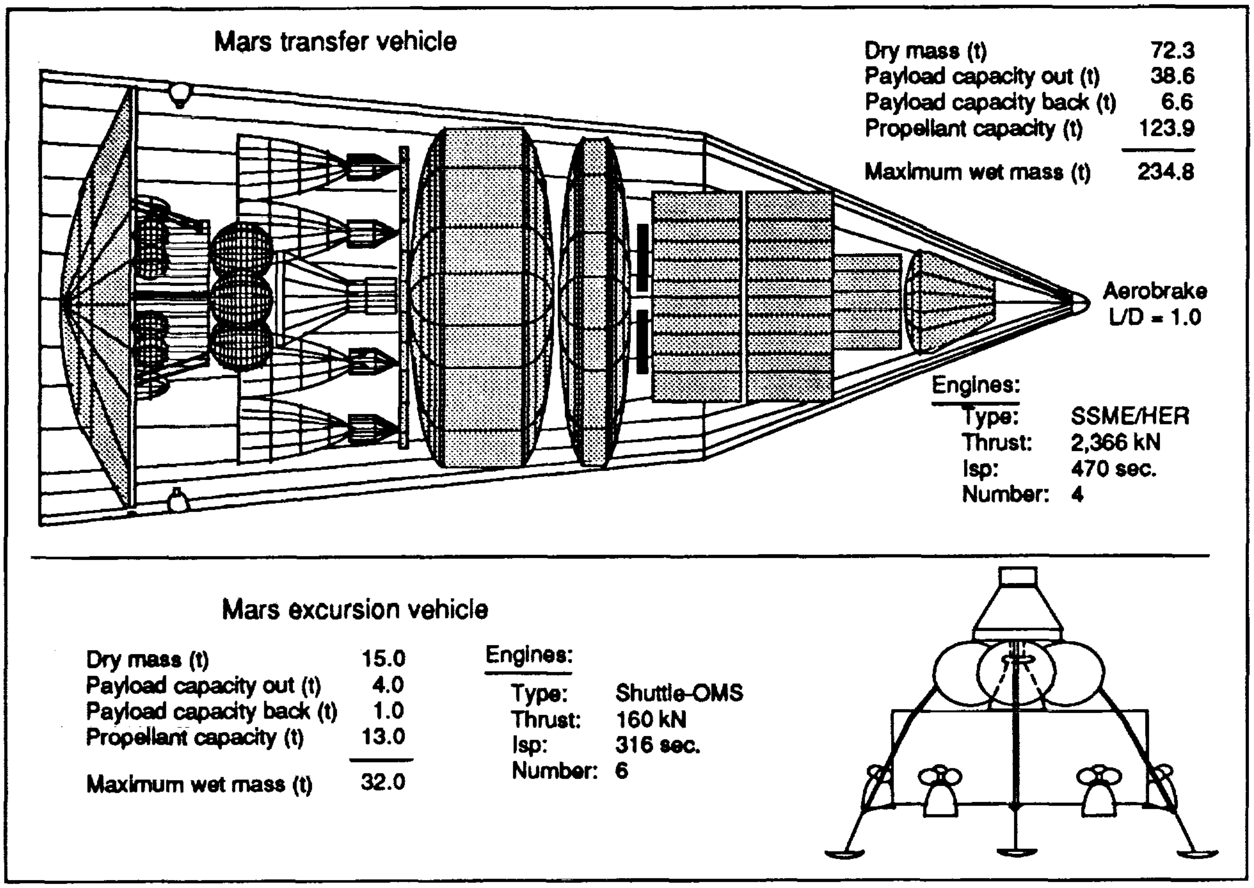


Figure 3.3.3-1.- Mars transfer and excursion vehicles.

(mass = 15.5 percent of total vehicle mass) for capture into a 500-km circular Mars orbit. Once in Mars orbit, the aerobrake is jettisoned. The MTV is designed to spend 30 days in Mars orbit.

The Mars descent vehicle (MDV), shown in figure 3.3.3-2, is deployed from the MTV and carries three crew to the Mars surface for 20 days. The MDV utilizes an aerobrake (mass = 5 percent of total vehicle mass) and stored bipropellants to land the crew and the Mars surface payload on Mars. The MDV also deploys a parachute to reduce entry speed. At the end of the Mars surface stay time, the crew and returning payload enter the Mars ascent vehicle (MAV), which is attached to the MDV, and transfer from the Mars surface to Mars orbit to rendezvous with the MTV. The MAV, which uses stored bipropellants, is shown in figure 3.3.3-3.

After the crew return to Mars orbit from the surface, they enter the MTV. The TEIS provides the energy for transfer from Mars orbit to Earth. Just prior to reentry at Earth, the crew enters the Earth crew capture vehicle (ECCV) and directly enters Earth's atmosphere.

3.3.3.2 Enabling Technology

A significant enabling technology for the Mars Expedition case study is the HLLV capability for the ETO flights. This case study assumes that a 140 t HLLV will be operational prior to the 2004 earliest launch date. Expansion of launch capability from the present STS system is expected during the establishment of Space Station Freedom, and assuming the availability of the 140 t HLLV does not seem unreasonable.

In addition to ETO enabling technology needs, several other enabling technologies for the MTV systems include the following:

a. A high reliability environmental control and life support system

b. Propellant tanks with low tankage mass factors and low boiloff rates to reduce mass requirements in LEO

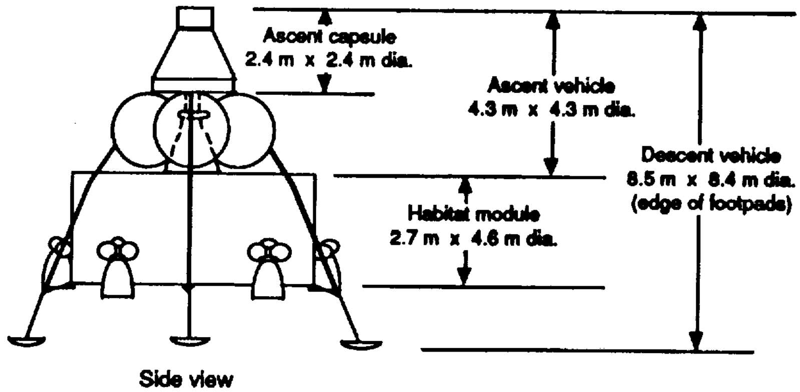
c. Propellant transfer in LEO capability

d. Remote rendezvous and docking capability in low-Mars orbit

e. High lift-to-drag ratio aerobrake for Mars aerocapture

f. Mars landing and hazard avoidance systems

g. Short-range forecasting technology for warning of solar flares (needed for pre-EVA activities).



Payload mass	
(includes MLMM and equipment)	10,000 kg
Payload volume	
MAV	
(cone - 2.4 m dia., 2.4 m ht.)	6.3 m^3
HAB module	
(cylinder - 4.6 m dia., 2.7 m ht.)	44.9 m^3
Propulsion system, descent	
Propellant type	MMH/N_2O_4
Engines	
number	6
type	Shuttle-OMS
mass (each)	134 kg
thrust (total)	160 kN (36 klbf)
Isp	3.1 kN-s/kg
Propellant mass	6,600 kg
Tank mass	200 kg
Terminal T/W	0.55 g
Total mass	35,200 kg

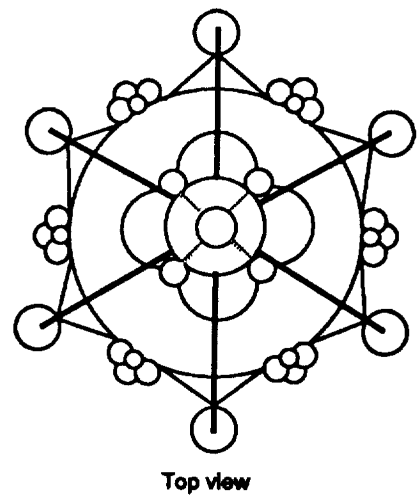
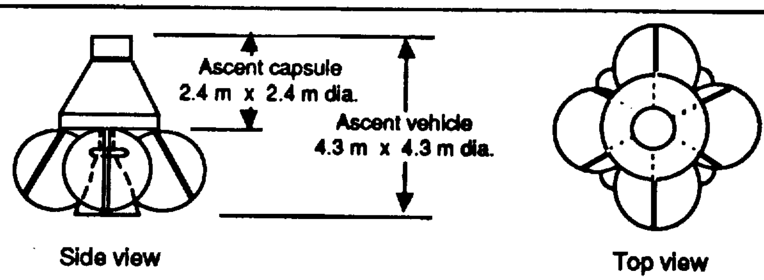


Figure 3.3.3-2.- Mars descent vehicle (MDV).



Payload mass to LMO	2,100 kg
Payload volume	
(cylinder - 2.4 m dia., 2.4 m ht.)	3.6 m^3
Propulsion system	
Propellant type	A-50/N_2O_4
Engines	
number	1
type	Apollo-CSM
mass (each)	373 kg
thrust (total	91.3 kN (20.5 klbf)
Isp	3.08 kN-s/kg
Propellant mass	8,690 kg
Tank mass	340 kg
Initial T/W	0.69 g
Total mass	11,163 kg

Figure 3.3.3-3.- Mars ascent vehicle (MAV).

3.3.3.3 System Alternatives

The split/sprint mission was specified in the SRD, and the major relevant trade is the all-up versus split/sprint approach. Mass in LEO savings were actually achieved by using the all-up strategy. This savings occurred for two reasons: first, the decision was made, from a crew/mission safety point-of-view, to remove the fueled TEIS from the cargo vehicle and place it on the piloted (sprint) trajectory. This decision was made because it was felt that the Nation would be unwilling to send a piloted crew to Mars without also sending along their means of return to Earth with them. Once the large mass TEIS is removed from the cargo flight and put on the piloted flight, the mass advantages of splitting the missions become very questionable. Second, the so-called "sprint" trajectory at last year's studies had a round-trip flight time of 440 days. By using an increased round-trip flight time of 500 days, the TMI propulsion requirements are significantly reduced, almost to the point of a conjunction-class mission. The two factors, when coupled together, actually give the all-up approach a mass in LEO benefit over the split/sprint (825 t for split/sprint versus 776 t for all-up).

3.3.4 Orbital Node Systems

The ground rules for the Mars Expedition case study specified that no orbital assembly would be required. Therefore, the case study requirements of Space Station Freedom were limited to providing the capability to qualify life support systems for long-duration flights and access to Freedom's existing capabilities, provided that the Mars Expedition requirements are compatible with Space Station Freedom.

3.3.5 Planetary Surface Systems

The surface elements to support a Mars Expedition respond to system requirements and mission objectives. Since this case study assumes a single mission with limited surface stay time, the requirements are less stringent than those for the Mars Evolution case study.

3.3.5.1 Elements and Systems

Habitation. Since the crew's stay on the martian surface will be only 20 days, there is no need for a dedicated habitation facility. During the expedition, the crew resides in the Mars excursion vehicle (MEV), specifically the descent portion of the vehicle.

EVA Systems. The exploration of the landing area necessitates frequent EVAs for purposes of deploying scientific instruments, collecting samples, scouting the terrain, and documenting the occasion. An extravehicular mobility unit (EMU) consisting of a pressure suit and portable life support system (PLSS) developed specifically for the martian environment is central to the EVA system. EVA systems are further enhanced with Shuttle-type hand tools and an unpressurized rover for local area access.

Life Support Systems. Life support for the crew while performing intravehicular activities on the surface is provided by the MEV. As stated above, EVA life support is managed by the PLSS associated with the EVA systems.

Thermal Control Systems. Like the life support systems, the thermal control systems for surface habitation are provided by the MEV, whereas the EMUs contain internal thermal control mechanisms.

Radiation Protection. The inherent protection offered by the martian atmosphere prevents the galactic cosmic rays from being a significant radiation threat. Artificial radiation shielding is not deemed necessary.

Power. An external power source is not required for the expedition operations. The MEV supplies power for habitation support while the crew is on the surface.

Assembly and Construction. The operations defined for the 20-day Mars Expedition do not require assembly or construction of surface infrastructure.

Surface Transportation. In order to enhance the exploration capability of the three-person team, an unpressurized rover is manifested as surface support equipment. The rover provides crew mobility for scientific equipment deployment, including meteorological balloon launches and atmospheric station set-up. The four-wheeled vehicle selected resembles the Apollo rover. The rover's payload capacity includes two suited crew and 500 kg of payload or three crewmembers with a lesser payload.

In Situ Resource Utilization. The objectives of the Mars Expedition case study do not include utilizing the local resources.

Launch and Landing Services. The Mars Expedition mission consists of a single MEV landing at the Ganges Chasma. The landing vehicle is characterized as a self-contained, expendable vehicle with both ascent and descent stages. All launch and landing requirements, whether navigation or thermal control, are met by the MEV hardware.

User Accommodations. The accommodations offered to the science community are basically all the surface systems selected: EVA systems, unpressurized rover, and assembly hand tools.

Telecommunications, Navigation, and Information Management. Dedicated communications equipment is not considered for surface support. Instead, communications capabilities are built into certain systems. Such systems include the lander, unpressurized rover, the EMUs, and remote science instrumentation. Separate equipment for surface communications is not required.

3.3.5.2 Technology Needs

The transient nature of the Mars Expedition surface systems means that technological advancements are unnecessary in many areas. Two areas, however, are identified where advancements in technology would provide improved technical performance: dust contamination control and EVA systems. A method of easily and effectively controlling and removing dust from EMUs, airlock seals, and other systems is required.

To meet the EVA requirements of the Mars Expedition, martian EMUs and portable life support systems must be developed. Technology development must emphasize minimal expendables, mass, and volume. A critical technology need is reduction of consumables, as transportation of consumables directly reduces the amount of other payloads that can be delivered.

3.3.5.3 System Alternatives

Many studies were carried out to select the surface systems used in this case study. In Volume III of this Annual Report, Section 5 (Distributed Studies of Equipment and Subsystems) discusses in detail the designs and characteristics of planetary surface systems.

3.3.6 Synthesized Mission Manifest

The Mars expedition mission requires the following major systems:

a. Interplanetary piloted vehicle

b. Descent vehicle

c. Ascent vehicle

d. Trans-Mars injection stage

e. Trans-Earth injection stage

A complete manifest is provided in figure 3.3.6-1. The first ETO flight transfers the piloted vehicle without the TEI and TMI stages attached. The next five ETO flights transfer the TEI stage, four TMI stages, and propellant. The Shuttle launch carries the three-person crew.

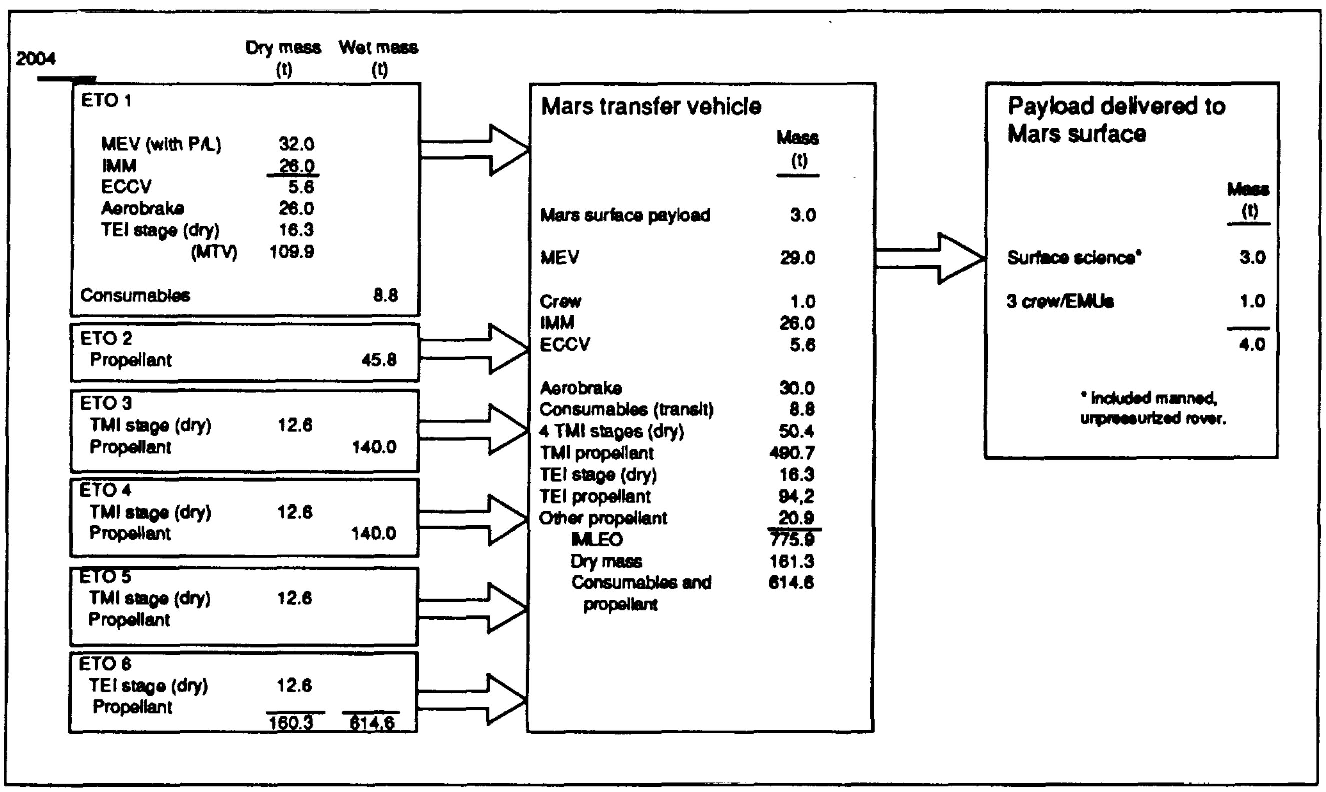


Figure 3.3.6-1.- Mars Expedition manifest (500 km circular Mars orbit).

3.4 CASE STUDY SUMMARY

The FY 1989 case studies continued the process of assessing approaches and alternatives for expanding human presence into the solar system. The case studies described in the previous sections were developed to identify issues for further study, and do not represent recommended implementations. For example, the use of on-orbit assembly for this year's case studies assessed: (a) no assembly (Mars Expedition), (b) assembly at Space Station Freedom (Lunar Evolution), and (c) assembly at a free-flying fixture (Mars Evolution). The purpose of each was to determine the implications of the concepts and was not intended to recommend one option over the other.

This year's Mars Expedition case study complements last year's Human Expedition to Phobos and Human Expeditions to Mars case studies. The primary objective of both studies was to define the requirements for sending humans to the martian system at the earliest feasible date, and the nuances of landing either on Phobos only, or proceeding directly to the Mars surface. Last year's Lunar Outpost to Early Mars Evolution case study assessed an evolutionary exploration approach by first establishing a lunar outpost and then proceeding on to developing a Mars outpost. However, last year's study did not consider the lunar element as a separate, permanent evolving outpost. This year's effort explored this concept in more detail by studying two separate evolutionary paths, one for the Moon and one for Mars. Both case studies evaluated an evolving outpost in its entirety.

As stated before, none of the case studies should be considered an exploration strategy proposal. Rather, it was the goal of the FY 1989 exploration studies to develop a database from which subsets, or individual pieces, from the different case studies could support a defined space exploration approach. However, certain basic components of an exploration initiative are independent of the target or mission sequence, such as the need for a heavy lift launch vehicle, the need for expanded on-orbit operations, and the need to resolve a number of life sciences issues. A second goal of the case studies approach was to establish requirements for the identification and resolution of these independent issues.

SECTION 4

Supporting Infrastructure Description

All human exploration program flights (and/or missions) will originate, terminate, and be supported on Earth and/or in Earth orbit. In addition to the technological and scientific capabilities needed to expand human presence beyond Earth orbit, the human exploration program will require a supporting infrastructure to provide capabilities and services for establishing and maintaining a functional link to Earth. Significant compatibility and synergism between human exploration program plans and those of the NASA programs/projects that provide the supporting infrastructure are essential.

The exploration program supporting infrastructure will provide the required Earth-to-orbit and Earth-orbital transportation services, permanently manned Earth orbit facilities and services, and TNIM (telecommunications, navigation, and information management) support services as appropriate. The following sections describe the plans for the development of this infrastructure.

4.1 EARTH-TO-ORBIT TRANSPORTATION

This section summarizes the Earth-to-orbit (ETO) transportation requirements for the exploration case studies. The systems to support these case studies may not represent the optimum solution from the perspective of overall national needs; however, they provide a basis for identifying many issues and trades associated with ETO transportation. Numerous Earth-to-orbit launch vehicle concepts can support the FY 1989 case studies. These concepts range from current systems (Space Shuttle, expendable launch vehicles), to derivations of current systems (Shuttle-C, Shuttle-Z), to new classes of launch vehicles (advanced launch system (ALS), heavy lift launch vehicle (HLLV)). The actual launch vehicle(s) used for human exploration could be any one of these or a combination of them. The launch/on-orbit processing study (see section 6.7) provides additional information on selection of the proper vehicle/vehicle mix to support the exploration missions.

4.1.1 Launch Vehicle Characterization

The SRD requirements for all three case studies stipulated that the Earth-to-orbit transportation system be capable of lifting at least 140 t of cargo to a Space Station Freedom compatible orbit. This system must also accommodate payloads as large as 12.5 meters in diameter and 25 meters in length. This payload includes mission cargo, vehicles, and propellant. Large payload shrouds are required for delivery of the reusable space transportation vehicles, whereas smaller shrouds are sufficient for cargo and propellant deliveries. In addition, delivery to low-Earth orbit of the space transportation vehicles occurs less frequently, as compared to cargo and propellant delivery, due to the multiple mission life of the reusable systems.

Figures 4.1.1-1 and 4.1.1-2 show some of the Earth-to-orbit launch vehicle concepts considered. Choosing the optimum ETO transportation system involves numerous trades and considerations. These include, but are not limited to, the ETO delivery requirements, flight rate, on-orbit assembly capabilities, ground and space operations, current flight systems and technologies, resource requirements, and launch facility requirements. The Earth-to-orbit vehicle assumed for detailed analysis is designated HLLV for generic heavy lift launch vehicle. This vehicle differs from those utilized in the MASE-developed integrated mission descriptions for the Lunar and Mars Evolution case studies (see sections 3.1 and 3.2). The HLLV may require larger initial costs but provides a good starting point for driving out the trades associated with ETO transportation. In addition, this single vehicle satisfies the initial SRD requirements for all three FY 1989 case studies. The characteristics of the HLLV, its associated ground operations, and its facility requirements are discussed below.

HLLV Description. The vehicle chosen as the baseline ETO vehicle is referred to generically as a heavy lift launch vehicle (HLLV), and is illustrated at the far right in figures 4.1.1-1 and 4.1.1-2. (This vehicle is discussed in detail in NASA Technical Memorandum 86520, *Heavy Lift Launch Vehicles for 1995 and Beyond*.) The liquid rocket boosters (LRBs), called half-stages or substages, consist of four LOX/RP-1 tankset assemblies. At the base of each tankset, there is either a single or a dual engine arrangement; that is, the four tanksets incorporate two dual-engine and two single-engine sets of Space Transportation Booster Engines (STBEs), for a total of six STBEs.

The four boosters are located at 90-degree separation angles around the first stage LOX/LH_2 tanks (the core). At the base of the core, there are five LOX/LH_2 Space Transportation Main Engines (STMEs) with two position, high-altitude, high-expansion ratio nozzles.

All 11 (five first-stage core and six half-stage booster) engines are ignited on the ground and are burned in parallel until booster propellant depletion and staging. The LOX/LH_2 core engine nozzle skirts are extended and continue to burn to propellant depletion at orbital insertion.

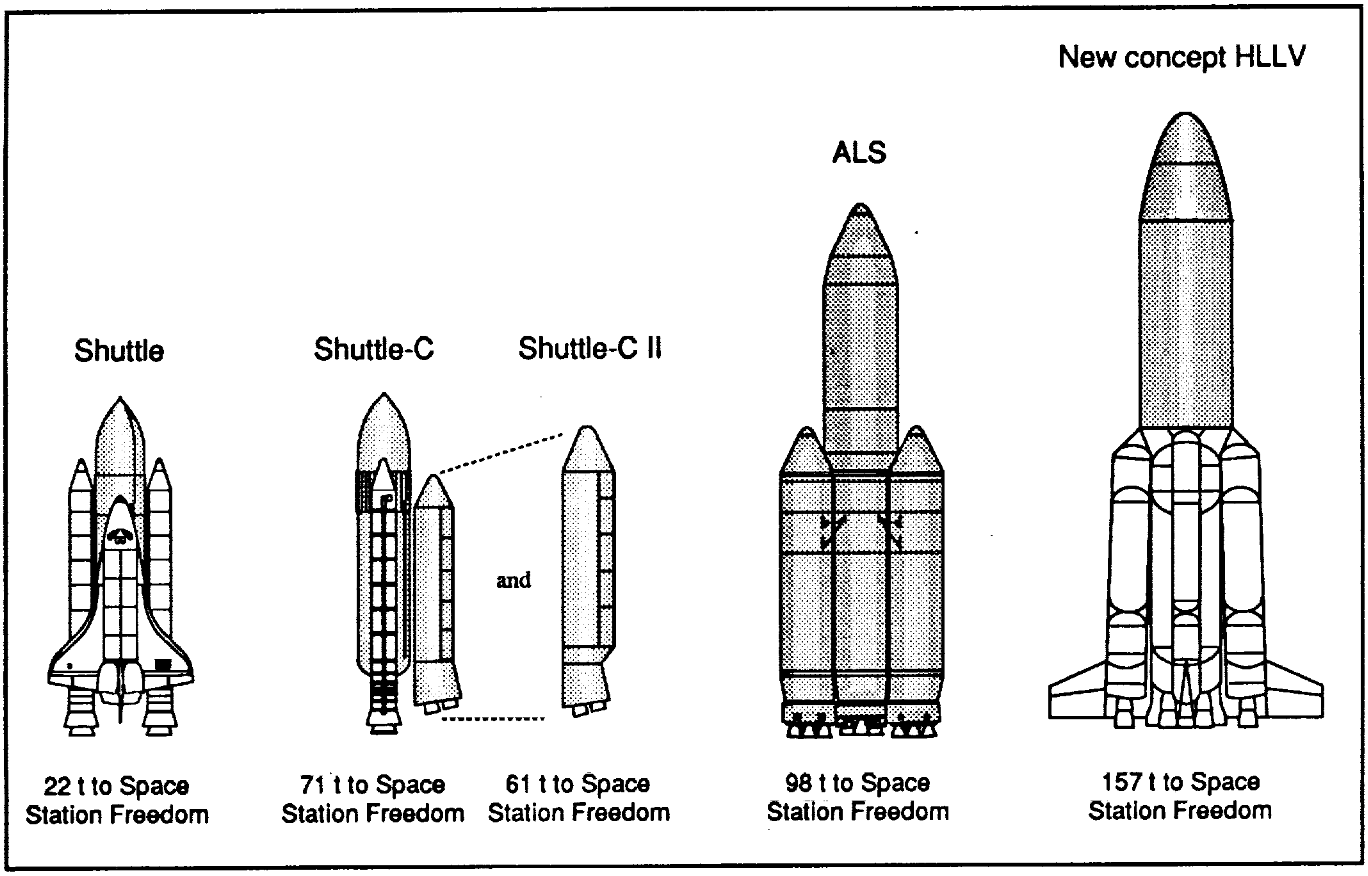


Figure 4.1.1-1.- Lunar Evolution Earth-to-orbit launch vehicle concepts.

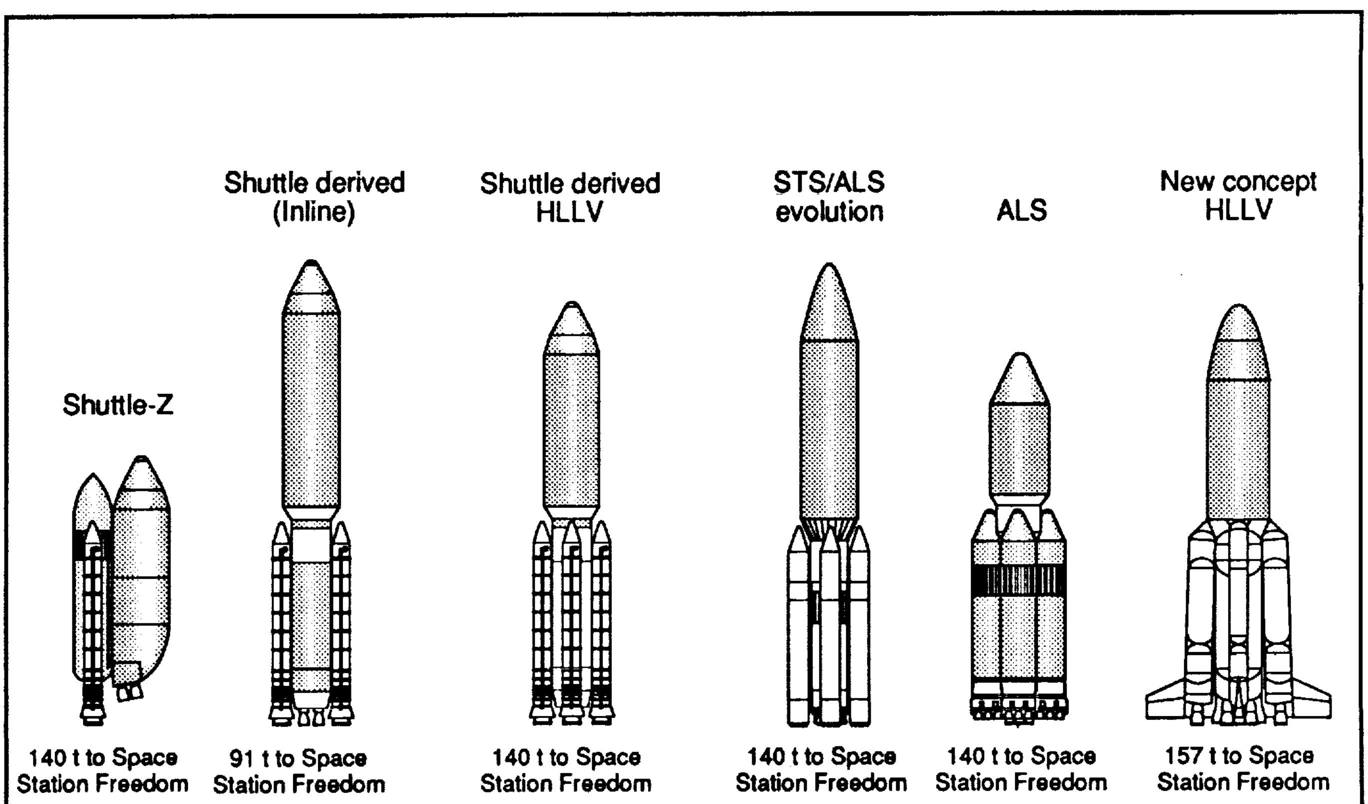


Figure 4.1.1-2.- Mars Evolution and Expedition Earth-to-orbit launch vehicle concepts.

The HLLV has a payload shroud 61 meters in length and 15 meters in diameter and will lift 157 t to the specified orbit. Both shroud size and lift capacity exceed SRD requirements, satisfying the required usable payload accommodations and providing a robust vehicle with a wide performance margin. The crew delivery requirements were assumed to be satisfied by the Space Shuttle.

Ground Operations. Because the HLLV is a new concept, its development provides an opportunity to improve launch-site ground processing. These improvements can be realized by incorporating ALS-type hardware technologies, advanced launch-site processing concepts, and a totally new infrastructure. The HLLV launch scenario incorporates the integrate/transfer/launch (ITL) approach with a central facility for vehicle and payload integration and a mobile launch platform (MLP) for transporting and launching the vehicle.

The operations flow for the HLLV is illustrated in figure 4.1.1-3. The components of the vehicle are delivered by barge to the launch site. The LRBs are processed in a horizontal processing facility (HPF), with two large processing areas; one is required for normal operations, the other provides a capability to accommodate surge requirements. After final checkout, each LRB is moved to the Vehicle Integration Building (VIB) and stacked on the MLP. After all four boosters are stacked, the core segment, which has been concurrently processed in the Core Assembly Building (CAB), is towed to the VIB and mated with the LRBs. Following payload checkout operations and payload/shroud integration, which are also conducted in parallel with other operations, the integrated payload assembly is moved to the VIB and mated to the core. At the conclusion of integrated testing and hypergolic propellant loading (if necessary), the vehicle is rolled out to the pad where final checkouts, cryogenic and RP-1 propellant loading of the vehicle, and propellant loading of the spacecraft/payload are performed prior to launch. Figure 4.1.1-4 shows projected timelines for these tasks.

Manpower required to support a peak HLLV launch rate of nine per year was estimated parametrically using the KSC Ground Operations Cost Model. This estimate, which assumes mixed fleet operations with an STS flight rate of 14 per year, indicates a need for approximately 4,000 people. The current STS requires the support of approximately 8,600 people.

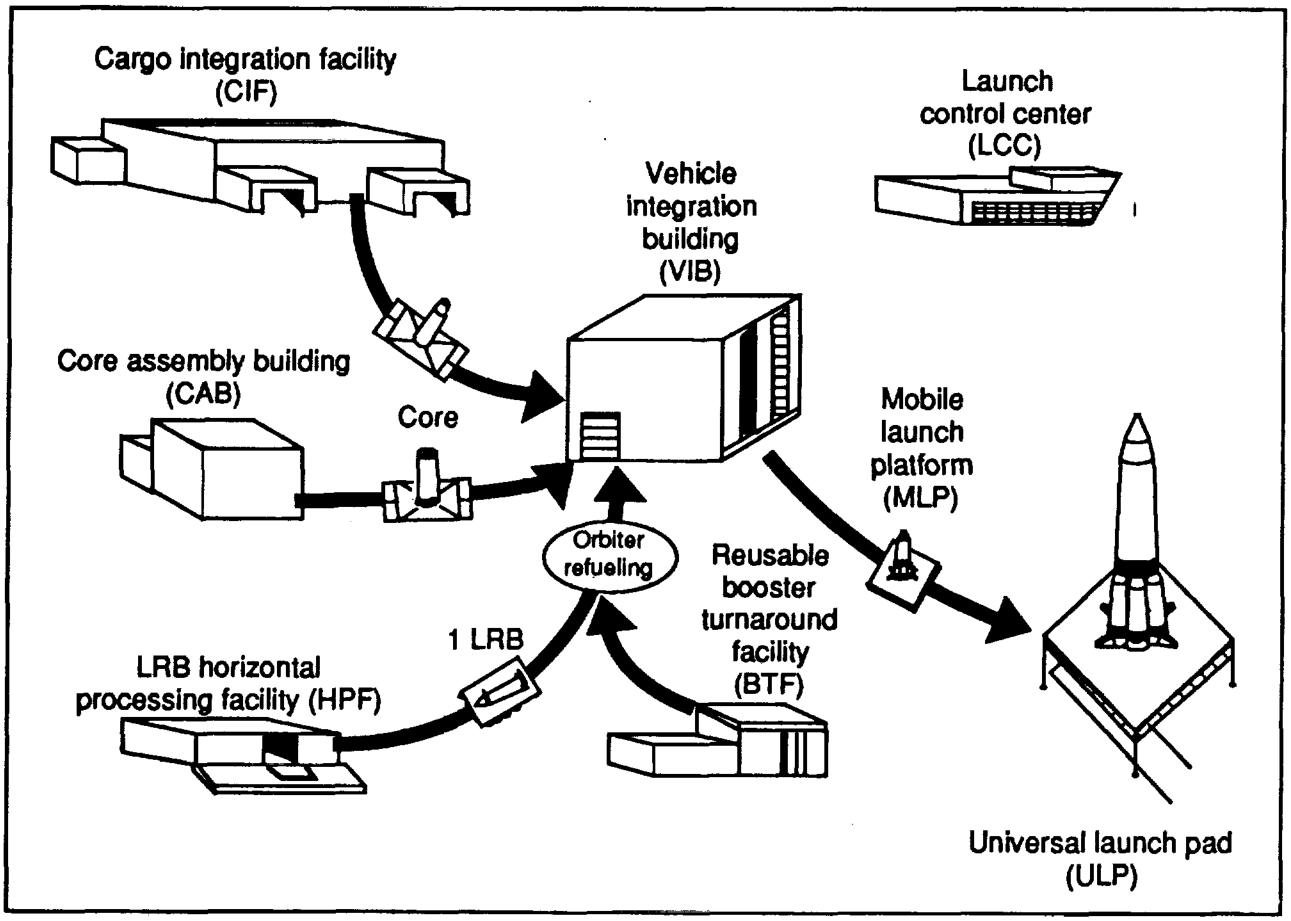


Figure 4.1.1-3.- HLLV processing flow (integration/transfer/launch).

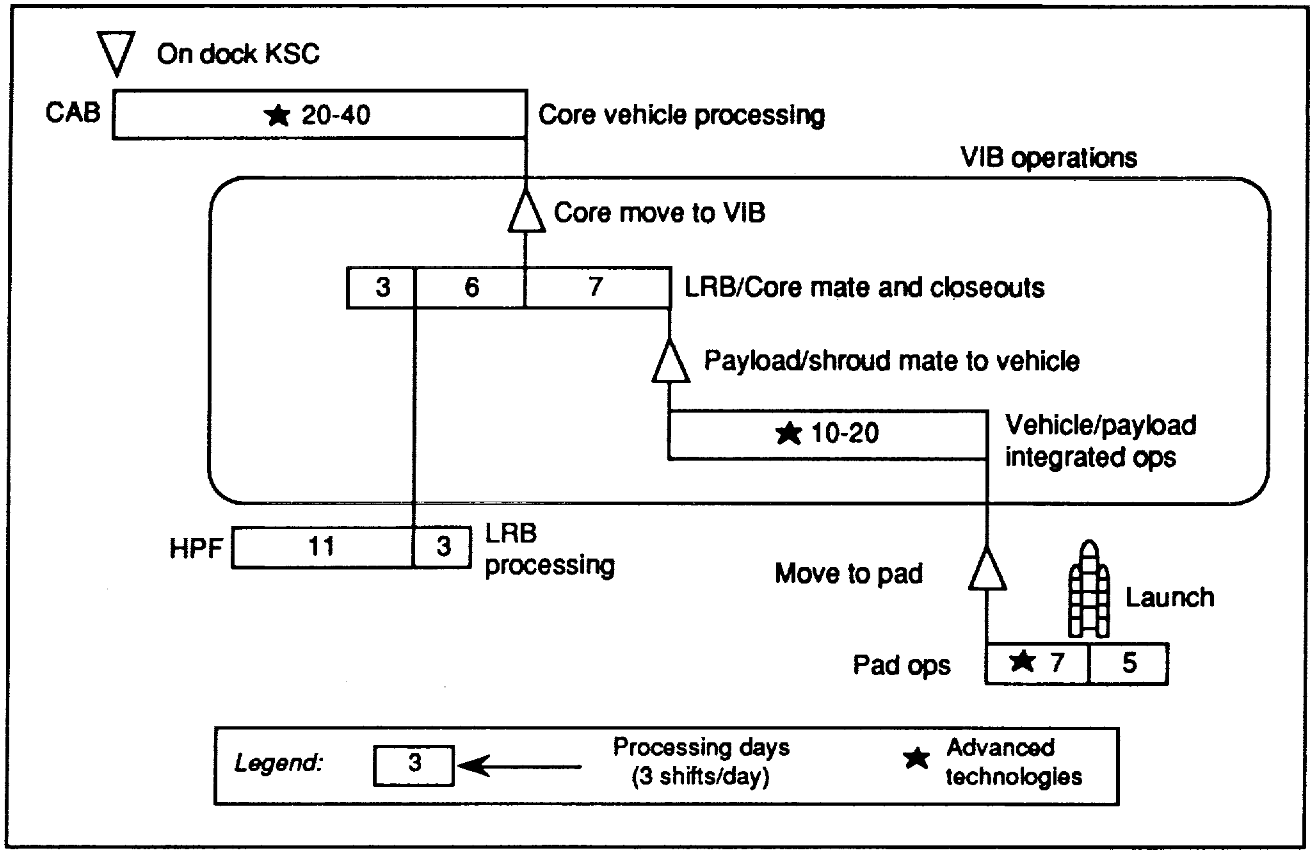


Figure 4.1.1-4.- HLLV processing timeline.

Facility Requirements. Two factors significantly impact the utilization of current launch facilities in support of the OEXP case studies. First, the size of the HLLV would make it impractical to utilize the existing STS facilities. Second, the lack of similarity between HLLV and STS flight elements would make redesigning the launch hardware for dual processing capability difficult, if not impossible.

A study ground rule stipulates that the HLLV facilities and ground systems must preclude a single catastrophic event from causing long-term disruption of ETO flight operations. Therefore, two universal launch pads (ULP) and two MLPs must be provided. A second VIB highbay is provided to assure ground processing capability in a multi-flow environment.

The facilities required to support the case studies with the stipulated launch vehicle and the total capacity of each facility are depicted in figure 4.1.1-5. Because the facility analysis assumes incorporation of advanced technologies in the flight and ground hardware, the HLLV processing systems afford a large growth potential.

The launch site development plan reflects an implementation for the introduction and integration of a new ETO launch vehicle (HLLV) into a mixed fleet environment. The plan assumes an FY 1992 decision to proceed, and consists of three non-autonomous phases: (1) activation, (2) transition, and (3) operations. The activation phase includes end-to-end implementation of the first line facilities required to support the ETO launch vehicle and initial launch capability (ILC), and the second line facilities that are required as the ETO flight rate ramps up. The operational phase extends over the program duration, beginning with support to the activation design development, continuing with staffing and training, and concluding with full operational capability supporting a sustained ETO flight rate.

4.1.2 Program Development

The HLLV development is expected to be similar to the development of the Saturn or STS launch vehicle systems. Development of the vehicle can be approached either directly or in an evolutionary fashion. A typical top-level schedule for direct development of a new launch vehicle system is illustrated in figure 4.1.2-1. It indicates that the earliest IOC would occur 8 years after authority to proceed. The evolutionary or derivative path approach would require a longer period with the actual development time depending on the selected evolutionary path. The HLLV launch site implementation schedules for the Mars Evolution and Mars Expedition case studies are shown in figures 4.1.2-2 and 4.1.2-3.

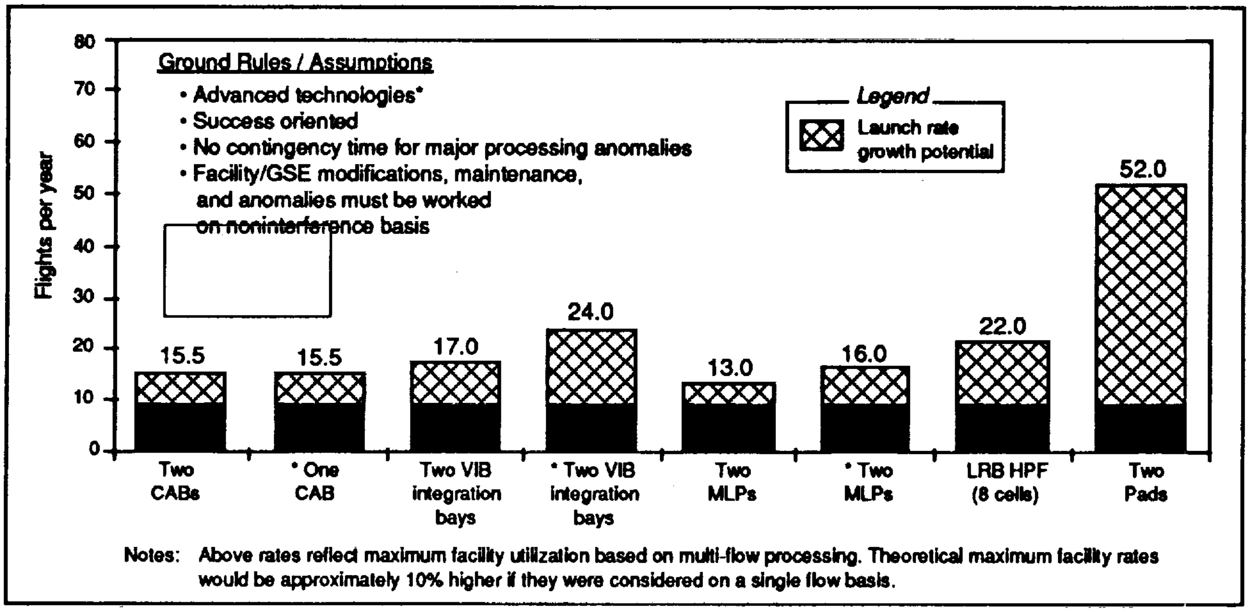


Figure 4.1.1-5.- HLLV facility capabilities beyond 2000.

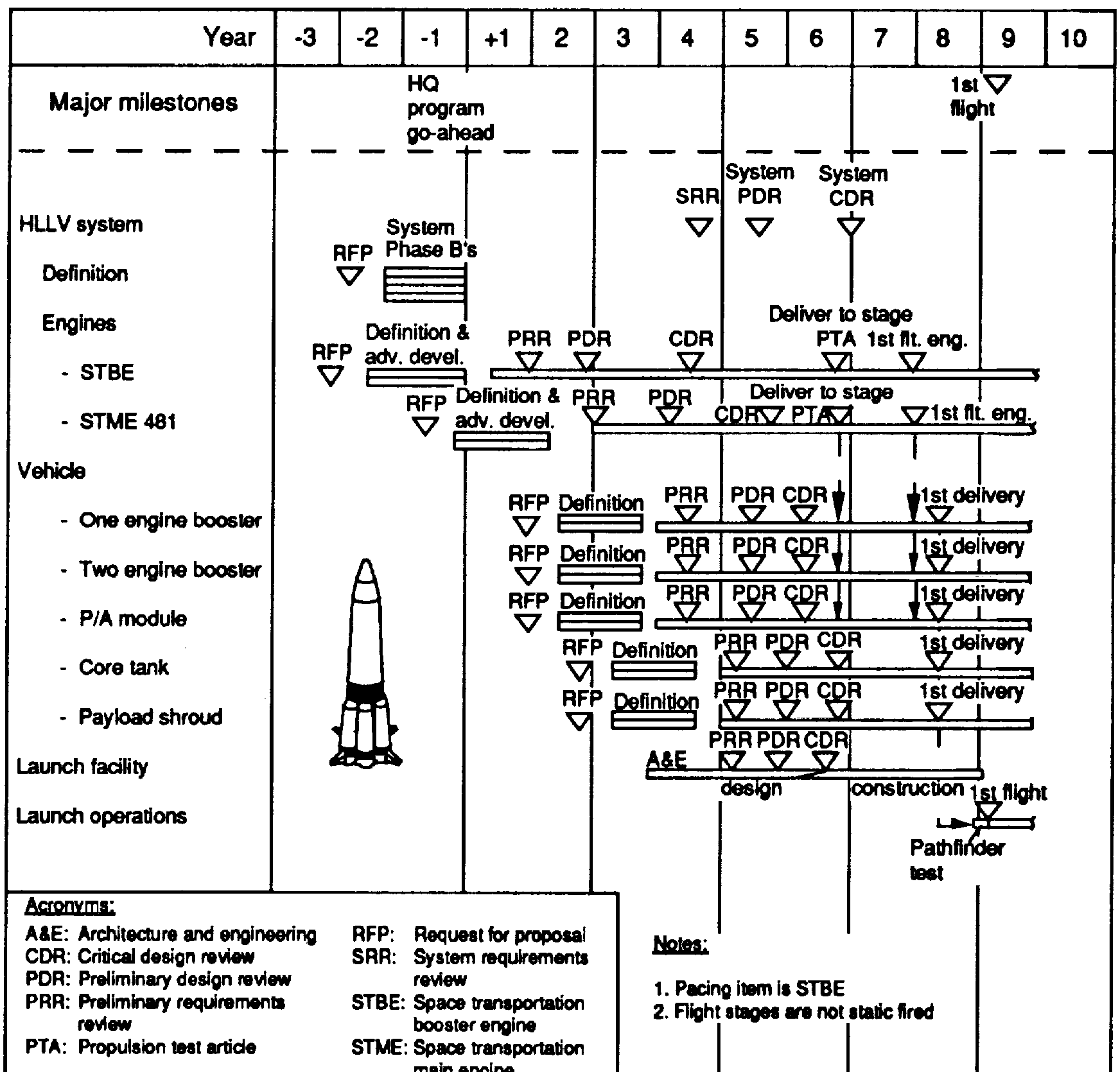


Figure 4.1.2-1.- HLLV development schedule.

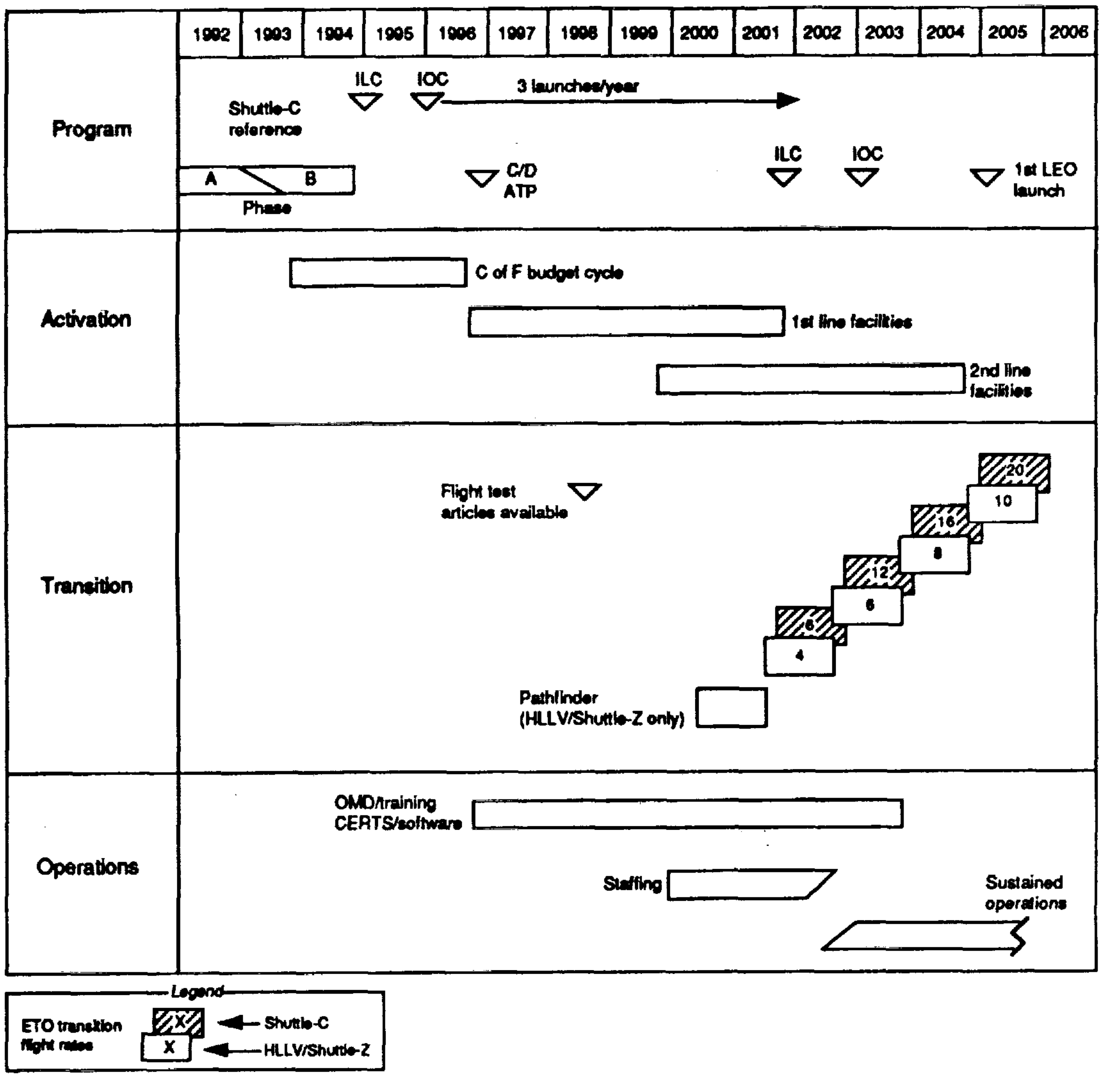


Figure 4.1.2-2.- Mars Evolution launch site implementation schedule.

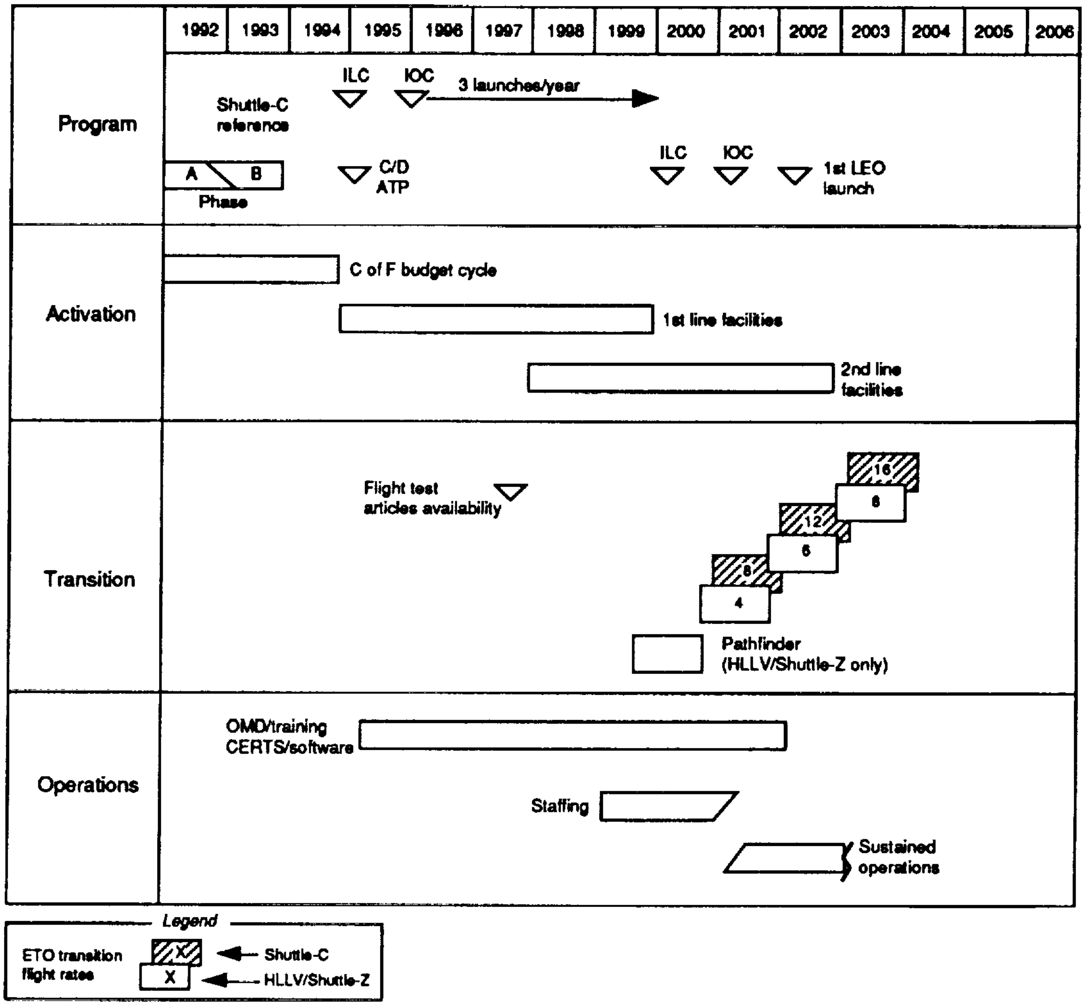


Figure 4.1.2-3.- Mars Expedition launch site implementation schedule.

4.1.3 Technology Development

A new launch vehicle program could take advantage of new and emerging technologies to improve both HLLV processing and flight hardware. Recent study programs, such as the Space Transporatation Architecture Study, Advanced Launch System (ALS), and the Space Transportation Infrastructure Study, have identified launch vehicle systems technologies with the potential for the greatest payback. For ground systems, these technologies include automated manufacturing processes, automated test and checkout, advanced facility design concepts, and automated or "paperless" administration. Vehicle technologies include advanced materials (such as aluminum lithium and advanced composites), advanced avionics, and automated diagnostics.

The technologies mentioned above are applicable to the HLLV; however, further definition of the technologies is required to determine realistic development schedules. As the availability dates of these technologies become better known, those compatible with the HLLV schedule can be targeted for incorporation into design.

4.1.4 Summary

A launch vehicle to support the OEXP case study requirements has been defined. The support operations, development schedules, and technology requirements have been identified. The analysis of the case studies shows that they are achievable, given a launch vehicle of the stipulated capabilities. The Lunar Evolution case study has the lowest schedule risk, but it may not even require a vehicle with the capability of the HLLV. The Mars Evolution case study has moderate schedule risk and requires delivery of large heavy payloads, but has the potential for long-term underutilization of the launch vehicle systems. The Mars Expedition case study has the highest schedule risk and would severely underutilize the launch assets if no follow-on missions were planned.

Issues that should be addressed in the future are:

ETO Vehicle Selection and Operational Scenario. The launch/on-orbit processing controlled trade study (discussed in section 6.7) has identified other vehicles and

vehicle combinations that may be more efficient in supporting specific mission needs. The Shuttle-C and a propellant tanker concept and the Shuttle-Z (or other derivatives) may be attractive alternatives for the Lunar and Mars Evolution case studies. Introducing and assessing the potential applicability of these vehicles will require better insight into on-orbit operations, propellant transfer, and propellant storage in space.

On-Orbit Transportation. In this analysis, all on-orbit transportation was accomplished by the orbital maneuvering vehicle (OMV). However, given the volume and mass of many planetary payloads, the OMV capability will require considerable upgrading. Future analysis is needed to develop more insight into the on-orbit transportation requirements.

Mission Requirements. This analysis has focused solely on the OEXP case studies, without considering other national mission requirements during the same time period. Future analyses, incorporating other civil and DoD mission requirements, will provide more insight into the true ETO needs during this time period. This analysis should identify the synergism between various sets of mission requirements.

4.2 SPACE STATION FREEDOM

4.2.1 Role of the Office of Space Station in Human Exploration

This section contains the Office of Space Station's (OSS) assessment of the impact of case study requirements on the Space Station Freedom program. Detailed assessments of the implications of supporting the OEXP case studies and implementation plans required to accommodate the various lunar and Mars transfer vehicles, orbital support equipment, payloads, and additional support infrastructure are discussed below. Preliminary approaches for implementing the exploration requirements include deriving systems concepts and developing formal growth requirements for the baseline Freedom Station. A small-scale advanced development program has also been inaugurated. Long-range planning mechanisms have been set in place to ensure that Space Station Freedom can co-evolve with the overall space infrastructure needed to support human exploration over the next 30 years.

To gain a better understanding of the relationship and interaction between Space Station Freedom and other program elements (e.g., ETO vehicles, lunar and Mars transfer vehicles, propellant storage/transfer methods), the use of Freedom was varied between case studies to range from heavy involvement (such as space-basing and processing of transfer vehicles) to minimal life sciences support.

It is important to note that the Space Station Freedom baseline configuration used in this section is consistent with the baseline as defined in the January 1989 to July 1989 timeframe. Therefore, this baseline configuration does not reflect any of the Freedom rephasing activities conducted after August 1989.

4.2.2 Space Station Freedom Transition Definition Program Support

A Transition Definition element has been established within the Space Station Freedom program to define and prepare for Freedom Station's evolution to meet the needs of users and long-term national goals. Preliminary plans that will satisfy eventual accommodation of exploration missions at Space Station Freedom address technical performance levels, schedule milestones, and budgetary requirements. Technical emphases will be placed on refinement of evolution concepts, definition and incorporation of baseline design accommodations (hardware "scars" and software "hooks"), and development of technology readiness to enhance the baseline and enable evolution of Freedom. The study plans in these areas are formulated to answer certain key questions fundamental for developing Freedom Station to serve as the transportation node for human exploration missions.

The milestone chart in figure 4.2.2-1 forms the basis of this planning. The chart reflects the fact that the Transition Definition program is the bridge between ongoing NASA planning and technology development programs and the Space Station Freedom development program. All the evolution milestones for the next 3 years are tied to the baseline program milestones.

The added functional capabilities required of a transportation node mean that the additional baseline design provisions must be identified to enhance and enable evolution. In this instance, evolution from the baseline can occur on-orbit through the addition of new hardware and/or the insertion of improved technology. The baseline preliminary design review (PDR) is realistically the last opportunity to change the design of Space Station Freedom; therefore, system impacts must be well understood by the time it is completed. Additions or changes made after PDR will most likely result in additional program costs and potential hardware element delays. Phased PDRs are distributed over nearly a year's time, with some systems PDRs (i.e., data management system) beginning in December 1989, whereas the Space Station PDR starts in mid-1990.

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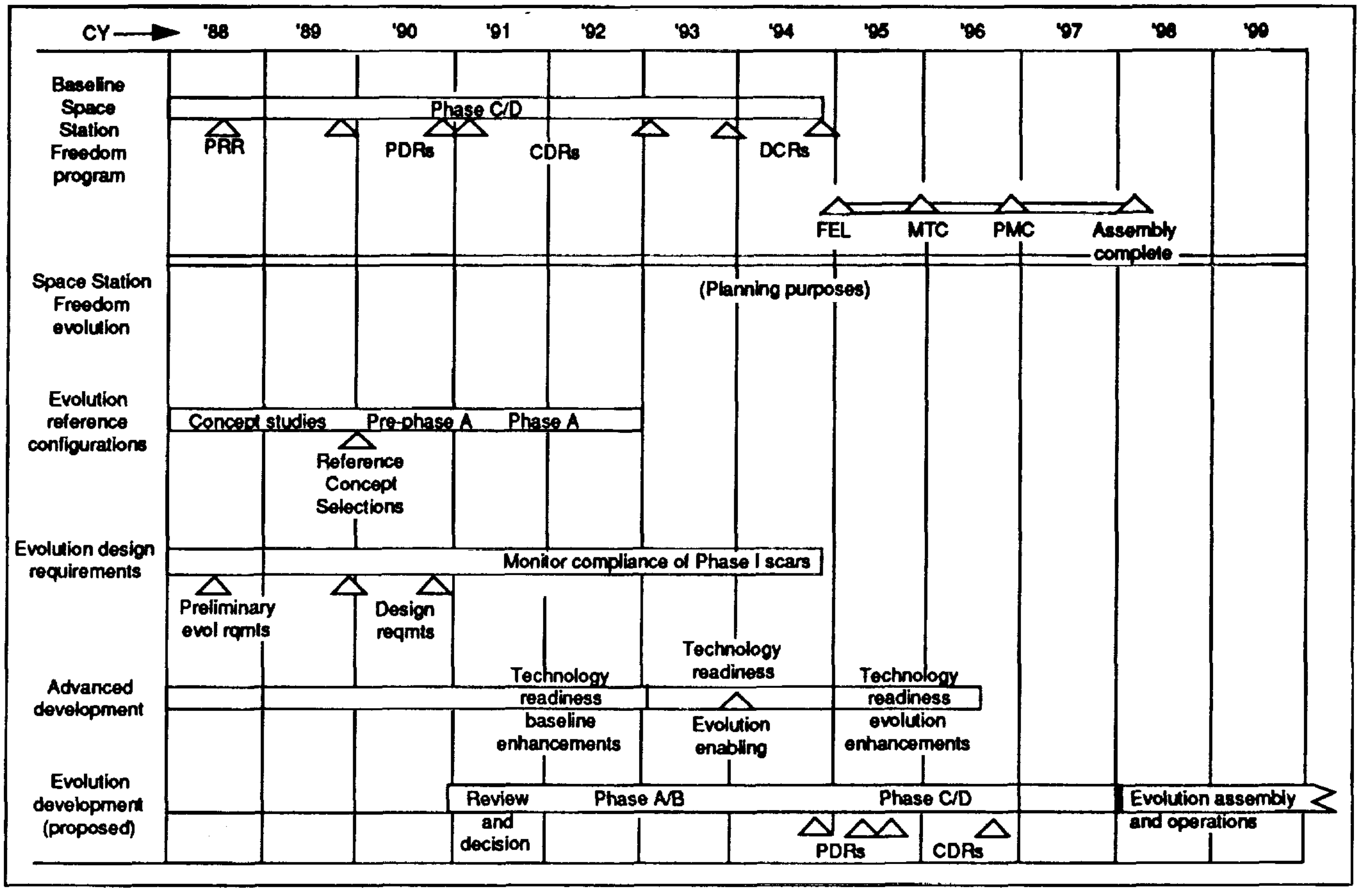


Figure 4.2.2-1.- Space Station Freedom evolution milestones.

4.2.3 Lunar Evolution Case Study

4.2.3.1 Case Study Requirements

A major utilization of Freedom's functional capabilities and resources was specified for the Lunar Evolution case study. The nature of the requirements (see table 4.2.3-I) will directly impact the evolutionary growth planning of Space Station Freedom. A detailed assessment of various growth concepts for both initial Lunar Evolution mission objectives and manifests and advanced mission requirements is provided below.

4.2.3.2 Space Station Freedom Growth

Growth configuration analysis (detailed below in section 4.2.3.3) defined the time-phased evolution of Space Station Freedom shown in table 4.2.3-II. The evolutionary growth is divided into two major phases that correspond with the evolutionary nature of the case study. The first phase, which is represented by growth deltas 1 through 6, can accommodate the first several expendable lunar transfer vehicle (LTV) flights. Once the LTVs become reusable, an increase in functional capabilities and resources will be required to store and process both the LTVs and their payloads. The operations, facilities, and resources required to accommodate the lunar mission elements (e.g., LTVs, payloads, mission crew, etc.) are discussed below.

Table 4.2.3-III shows the current resource requirement recommendations.

The growth rate is a direct function of available ETO lift capability. Figure 4.2.3-1 shows when the two evolution phases must begin to meet the stated mission objectives. In addition, a top-level assessment of the manifesting of ETO launches required to achieve the final growth delta is shown in figure 4.2.3-2 for both an Advanced Solid Rocket Motor enhanced Space Shuttle launch system (22 t to 407 km) and an expendable Shuttle-C launch vehicle (40 t to 407 km). As anticipated, more Space Shuttle flights are required to launch Freedom growth hardware than if a Shuttle-C launch vehicle were used. However, in this preliminary study, the Shuttle-C configuration chosen is underutilized in its available launch mass capability for a majority of launches. This result indicates that a detailed assessment of the design and manifesting of an alternative ETO launch system to the Space Shuttle is required to optimize ETO Space Station Freedom hardware launches.

TABLE 4.2.3-I.- LUNAR EVOLUTION CASE STUDY - SPACE STATION FREEDOM REQUIREMENTS

- Provide capability to support advanced development systems for lunar outpost and space transportation.

- Provide capability for housing transient mission crew, support crew, and mission equipment.
  - Transient mission crew: (Outbound) — Pre-experimental phase support mission crew of 4; Experimental phase and beyond support mission crew of 8
  - Transient mission crew: (Inbound) — Accommodate a crew of 8
  - Mission support crew: — Accommodate a support crew of 6, for 6-month tours of duty, twice per year
  - Mission equipment: — Accommodate 5 metric tons of cargo

- Provide space-basing accommodations to satisfy LEO traffic requirements for LTVs.

- Provide the capability to store lunar mission equipment awaiting transport to the lunar system. In addition Space Station Freedom shall provide the capability for mission equipment:
  - State of health monitoring
  - Assembly
  - Integration
  - Check-out
  - Preparation for transport by LTV
  - Other on-orbit processing, as required

- Provide capability to process space transfer vehicles on-orbit by supporting the following:
  - Vehicle mating/assembly and demating/disassembly
  - Space construction and elements of LTVs
  - Element and integrated vehicle on-orbit check-out
  - Maintenance and servicing of departed and returning lunar LTVs
  - Deployment and retrieval of lunar LTVs
  - Ground communications with LTVs (while berthed)
  - Provide housekeeping resources and services to LTVs and nodes
  - Loading and unloading of mission equipment from LTVs and/or ETO vehicles

  (Maximum vehicle processing time is currently 4 months.)

- Provide the capability to support on-orbit supply and resupply of:
  - Life support system fluids
  - Cryogenic and storable propellants
  - Mission equipment

  This includes providing for:
  - Fluid storage
  - Determination of fluid quantities
  - Fluid transfer interface capability
  - Operational control

- Provide debris protection for LTVs and mission equipment while resident at Space Station Freedom.

TABLE 4.2.3-II.- LUNAR EVOLUTION CASE STUDY - SPACE STATION FREEDOM GROWTH DELTAS

Δ1	Two 25-kW solar dynamic modules; two 25-meter transverse boom extensions; space-based OMV and space-based OMV accommodations
Δ2	Upper/lower keels and booms; utility trays
Δ3	One habitat module; two resource nodes
Δ4	Two 25-kW solar dynamic modules; customer servicing facility phase 1
Δ5	Customer servicing facility phase 2
Δ6	Phase 1 LTV processing facility; phase 3 servicing facility (completed CSF); one MSC; Shuttle-C docking adapter
Δ7	Additional lower keel and boom truss structure; utility trays
Δ8	Phase 2 LTV processing facility (LTV processing facility complete)

TABLE 4.2.3-III.- LUNAR EVOLUTION CASE STUDY - SPACE STATION FREEDOM RESOURCE REQUIREMENTS

Station Resource	Current Recommendation
Power	
Average	175 kW
Peak	NA
Crew	18
Pressurized modules	
US habitation	2
US laboratory	1
ESA laboratory	1
JEM laboratory	1
Pocket laboratory	0
Resource nodes	6
Transverse boom	
Dual keel	Scar for distributed systems
Servicing facility	Scar for facility and distributed systems
Power modules	Scar for addition of power modules at boom ends
Payloads	
APAE	18
Tether payloads	2

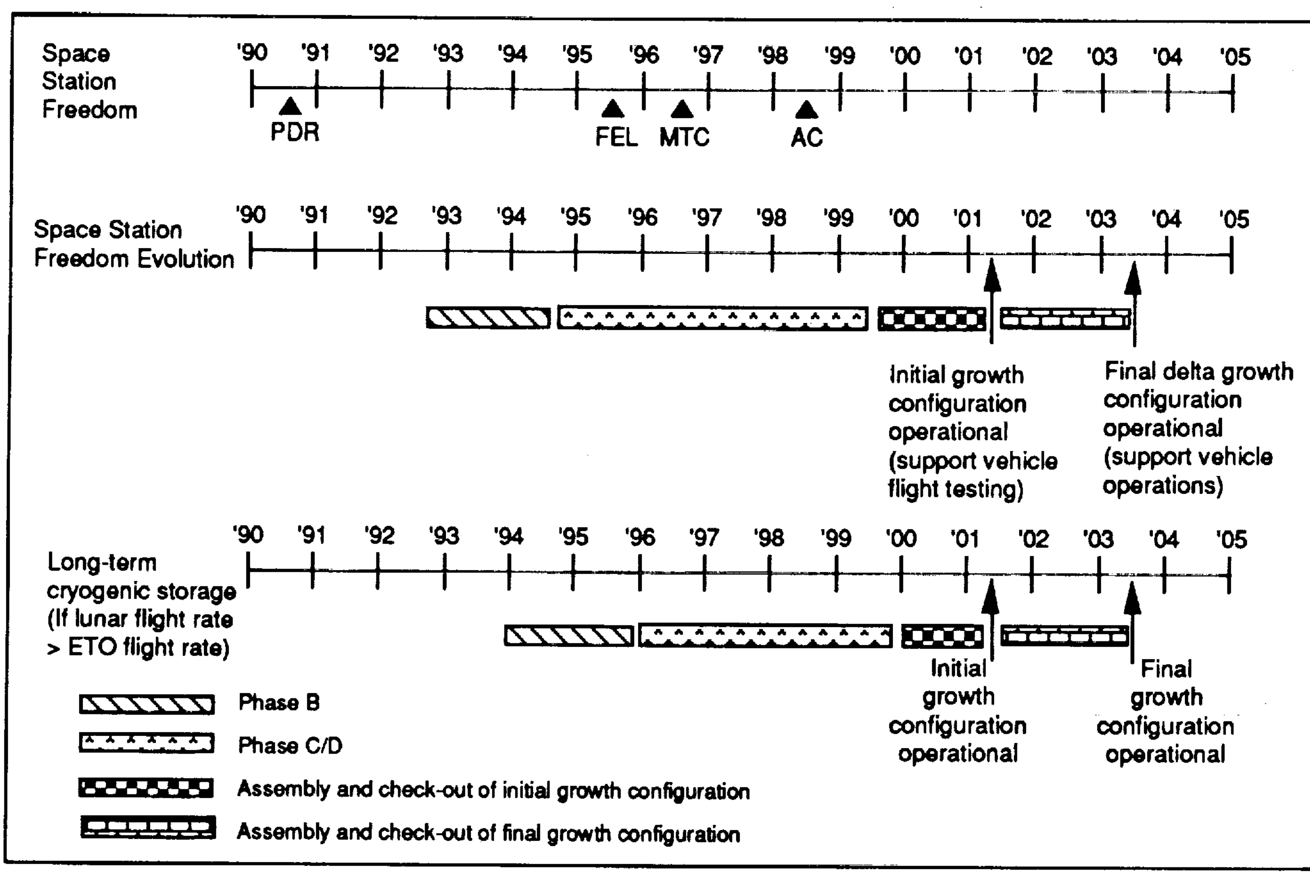


Figure 4.2.3-1.- Lunar Evolution case study: programmatic schedule for Freedom evolution.

Sequence \ Flight		1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17
Advanced rocket motor Enhanced Spaced Shuttle	Mass	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	○
	Volume	●	●	●	○	●	●	●	●	●	○	●	●	○	○	●	●	●
Expendable Shuttle-C	Mass	●	○	○	○	○	○	●	○	○	○							
	Volume	●	●	●	●	●	●	●	●	●	●							

● - Greater than 90% of available launch vehicle mass/volume utilized for cargo delivery

○ - Less than 90% of available launch vehicle mass/volume utilized for cargo delivery

Figure 4.2.3-2.- Earth-to-orbit manifest options for Space Station Freedom Lunar Evolution case study growth hardware.

4.2.3.3 Growth Configuration Analysis

Several candidate evolution configuration options, as shown in figures 4.2.3-3, 4.2.3-4, and 4.2.3-5, were analyzed using the 10 discriminators listed in table 4.2.3-IV. The most important discriminators were controllability, operability, and the static microgravity environment for each option. It is desirable to have an evolutionary Space Station Freedom that would serve primarily as a transportation node, and yet provide a quiescent research and development period when no major vehicle processing activities are occurring. To this end, configurations were sought that would provide all the functional capabilities of a transportation node (LTV processing, propellant storage and transfer, payload accommodation, etc.), minimize operational impacts on Freedom (controllability, reboost propellant, crew time, etc.), and still provide a suitable microgravity environment.

To determine which configuration(s) would be most suitable for each phase of evolution, the following analyses were conducted on each configuration:

a. Generation of solid computer models using IDEAS[2] software

b. Determination of mass properties including center of mass and moments of inertia

c. Open-loop control system sizing including secular and cyclic momentum determination and torque equilibrium angle (TEA) determination

d. Closed-loop control system sizing including secular and cyclic momentum determination and TEA determination

e. Calculation of microgravity envelope

f. Assessment of orbital decay and propellant reboost

g. Analysis of Space Station Freedom growth element and LTV/payload clearance

h. Determination of module cluster field of view

i. Determination of electrical power and thermal rejection

j. Characterization of structural dynamics including system modes and forced response (only on selected configurations)

The mass properties, open-loop control system sizing results, and growth hardware element tables are provided in detail in Volume IV of this annual report. To date, all the analyses used scenarios that assumed that either part or all of the mission propellant was stored directly on the LTVs or in a co-orbiting propellant facility. A further discussion of propellant storage implications is provided in section 4.2.3.4.

At present, only top-level comparisons and quantification have been conducted on and between the various evolution configurations. A detailed comparison is required, such as using the Multi-Attribute Utility Theory outlined in Volume IV of this series. Even so, a gravity-gradient stabilized configuration such as those shown in figures 4.2.3-4 and 4.2.3-5 is a more desirable configuration from a controllability perspective than configuration 1, shown in figure 4.2.3-3.

Configuration 1 was designed to locate the LTV close to the pressurized (habitable) region of Space Station Freedom to provide direct crew viewing during processing operations. In addition, if EVA is required, the crew would not have to travel a great distance from the air-

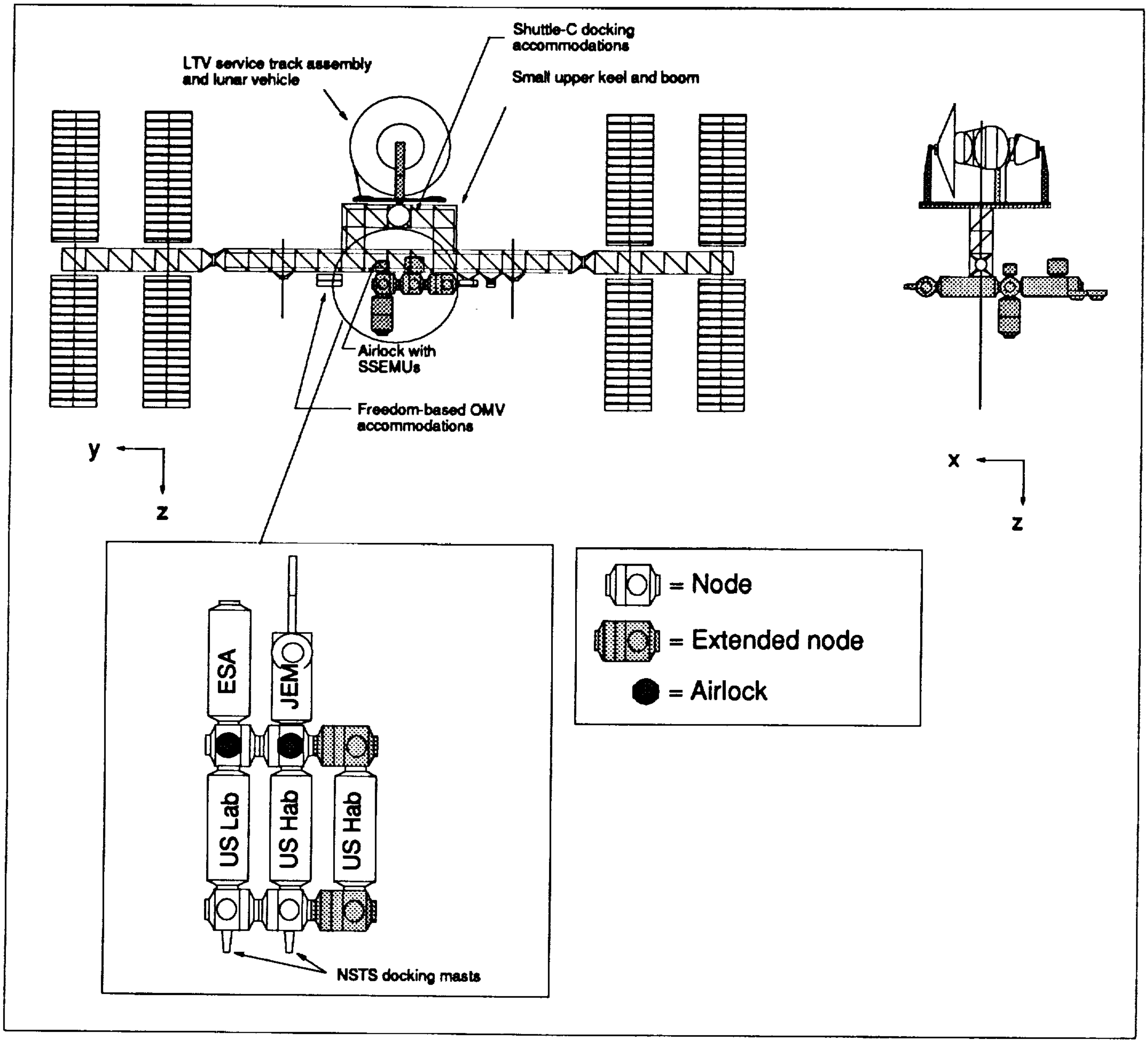


Figure 4.2.3-3.- Configuration 1: Accommodation at Space Station Freedom for LTV verification.

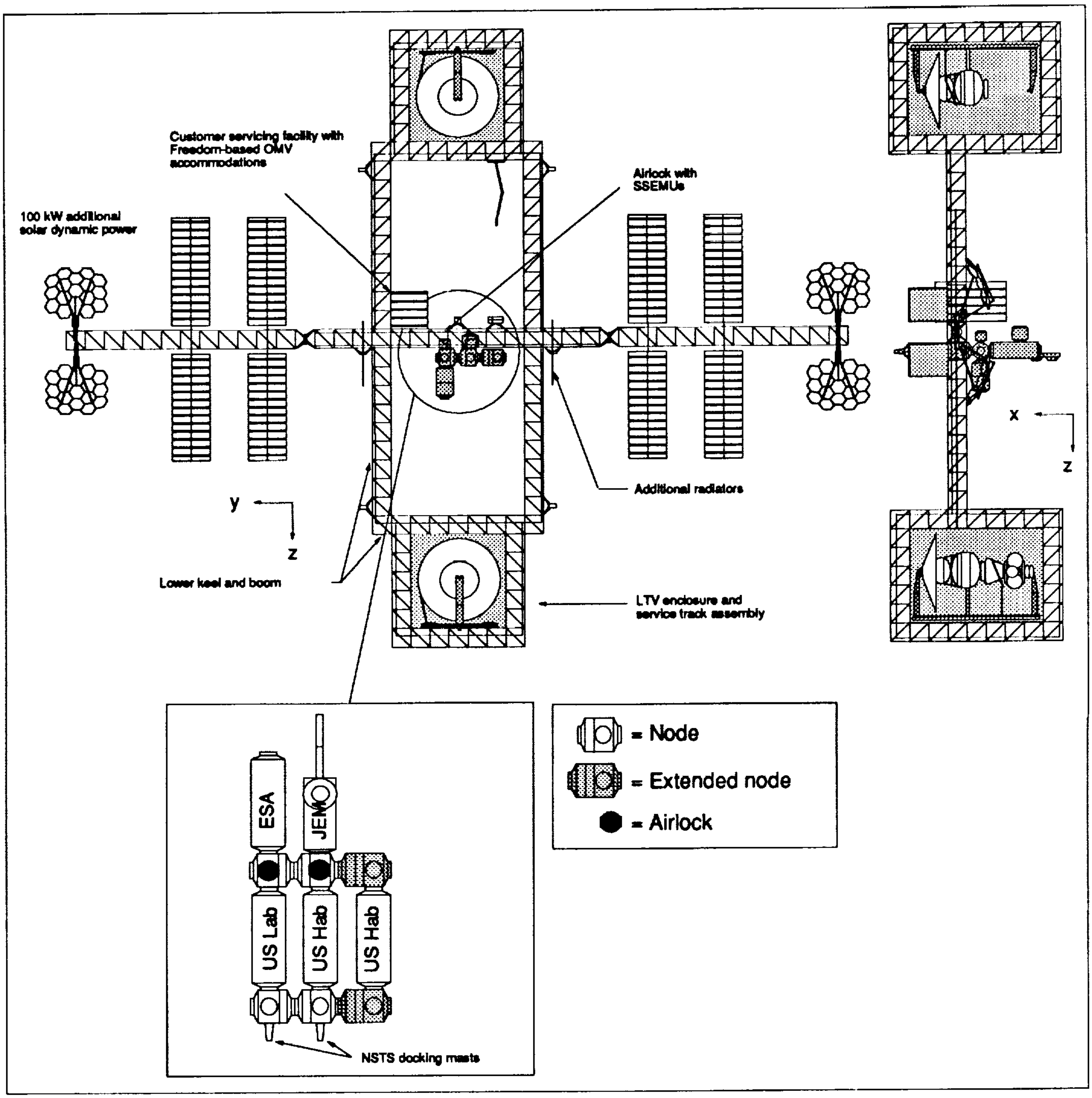


Figure 4.2.3-4.- Configuration 2: LTV accommodation at Space Station Freedom with two hangars.

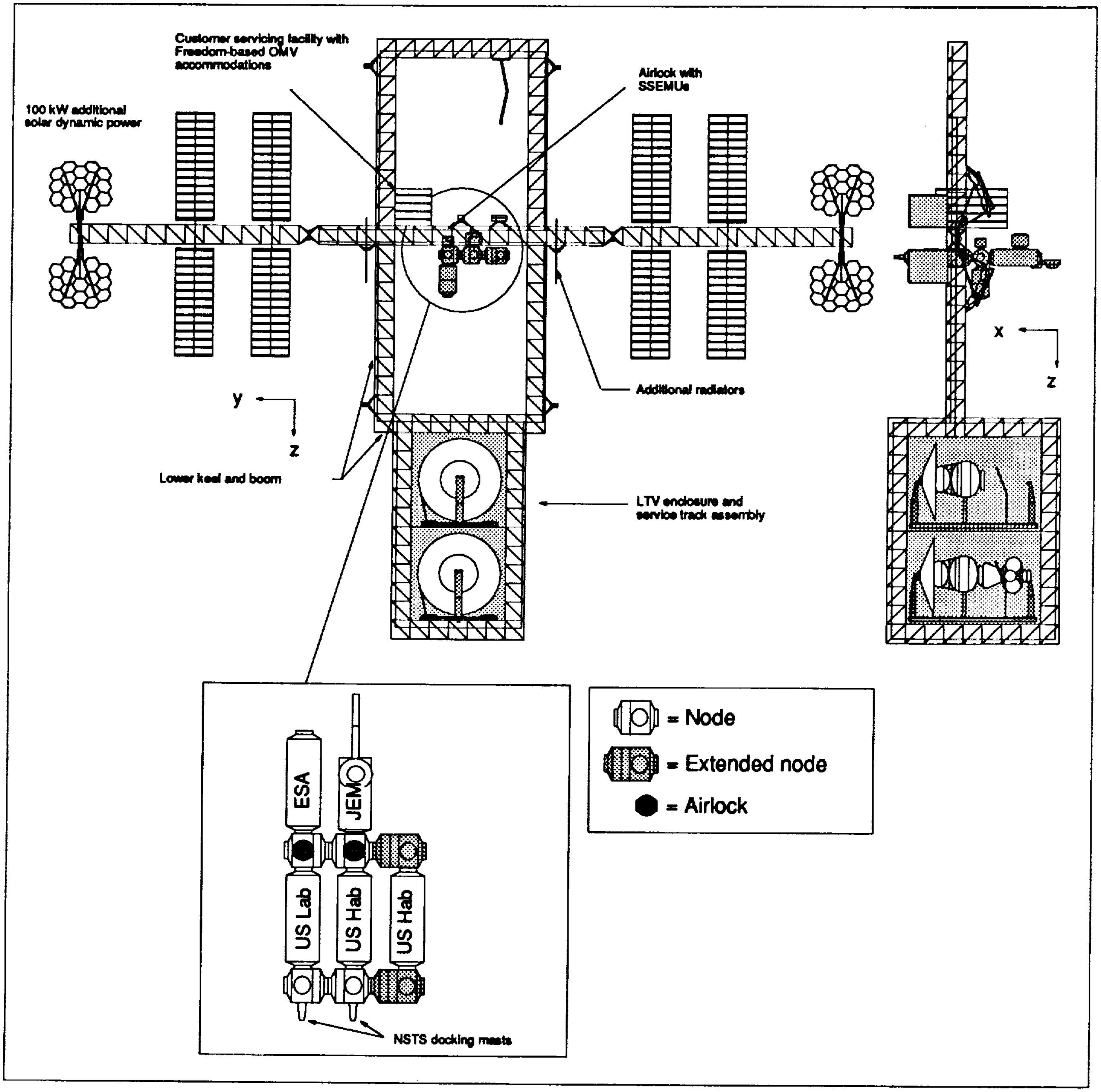


Figure 4.2.3-5.- Configuration 3: LTV accommodation at Space Station Freedom with dual LTV hangar.

TABLE 4.2.3-IV.- SPACE STATION FREEDOM GROWTH DISCRIMINATORS

Discriminator	Measures of effectiveness
Space Station Freedom control	TEA, CMG momentum storage requirements Roll magnitude to dump secular momentum
Structural capability	Truss member structural integrity under reboost loads Controls-structures interaction acceptability
Operational capability	Ability to perform all vehicle processing requirements
Microgravity environment	Highest microgravity level in the U.S. Lab module TEA (affects microgravity gradient along the Space Station Freedom x-axis)
Viewing	Stellar viewing from the JEM exposed facilities: Percent viewing blockage Greatest field of view Average swath angle
Main radiator interference	Clearance between radiator wing and growth elements
Module closure / dual egress	Number and length of time with modules having less than half closure (racetrack) or lack of dual egress (two ways out)
Structure / ease of assembly	Number of modules along the transverse boom Number of modules within MSC reach
Ability to accommodate further growth	Number of free and usable berthing ports on the final configuration
Growth mass	Total weight of growth elements (approximate measure of cost)

lock in the event that an emergency required an immediate return to Freedom. Unfortunately, this configuration would operate at an extremely large torque equilibrium angle (TEA) impacting Space Shuttle and Space Station Freedom docking/berthing operations. Also, the Freedom Station inertia properties lend themselves to requiring an increase in the number of control moment gyros required to accommodate the large increase in the momentum management requirements.

Configurations 2 and 3, shown in figures 4.2.3-4 and 4.2.3-5 respectively, have LTV processing facilities located away from the transverse boom on upper and/or lower keels and booms. This addition of keels and booms serves to provide a gravity-gradient stable configuration, reducing the cumulative environmentally induced momentum disturbances on Freedom. Although the LTV processing facilities are farther away from the manned modules, raising potential concerns for EVA safety, the separation of such processing activities as LTV fueling and payload mating from the pressurized modules provides an overall safer operating environment.

Finally, although configuration 2 provides a better static microgravity environment than configuration 3, it is better, from an operations standpoint, to have all the vehicle processing facilities and orbital support equipment co-located. Therefore, option 3 is recommended as the prime reference configuration to support the Lunar Evolution case study.

4.2.3.4 Propellant Storage and Transfer Options

Based on the data that have been generated on vehicle and payload mass properties, the greatest concern with accommodating the Lunar Evolution case study requirements is the storage and transfer of large quantities of cryogenic fluids. Table 4.2.3-V lists the currently understood vehicle mass and propellant requirements.

OSS, working with the Lewis Research Center, is presently investigating several different storage options. An overview of three potential on-orbit storage options is presented here, with a brief discussion of issues associated with each method. This topic also is addressed in section 6.7 of this volume, and an explanation is provided in Volume IV. In summary, the three options are:

Method 1: Propellant storage on Space Station Freedom

a. Propellant transfer from delivery system to LTV for storage

b. Propellant transfer from delivery system to Freedom for storage

c. Tank transfer from delivery system to LTV for storage

d. Tank transfer from delivery system to Freedom for storage

Method 2: Propellant tethered to Space Station Freedom

Method 3: Propellant co-orbiting with Space Station Freedom.

From a configuration assessment point of view, each of

TABLE 4.2.3-V.- LUNAR VEHICLE MASS PROPERTIES AND PROPELLANT REQUIREMENTS

Component	Dry mass (t)	Propellant (t)	Total mass (t)
Lunar transfer vehicle (LTV-C)	9.7	125.8	135.5
Transfer crew cab	9.0		9.0
LTV-P (LTV-C with a crew cab)	18.7	146.6	165.3
Lunar excursion vehicle (LEV-C)	3.4	24.9	28.3
Excursion crew cab	3.0		3.0
LEV-P (LEV-C with a crew cab)	6.4	24.8	31.2
LEV-P payload (a)	7.6		7.6
LEV-P payload (b)	14.9		14.9
LEV-P payload (total)	22.5		22.5
Crew	1.2		1.2
First piloted flight (LTV-P, LEV-P, LEV-P P/L, crew)	48.8	171.4	220.2
Standby cargo vehicle (LTV-C)	9.7	(dry)	9.7
Operational piloted flight (LTV-P, LEV-P P/L, crew)	43.4	160.5	214.8

the storage and transfer options listed under method 1 above will have almost the same impact on Space Station Freedom controllability and operational impacts.

At present, two different tether cases are being analyzed to determine their impacts on Space Station Freedom. The first case, illustrated in figure 4.2.3-6, uses a tether to store and transfer propellant for lunar mission flight 1, which is an expendable cargo flight. The second case uses a tether system to store and transfer propellant during the lunar utilization phase, in which two reusable LTVs are based at Freedom.

The third method of storing propellant would be to provide an OMV-type vehicle that would rendezvous with each arriving ETO propellant delivery tanker and maintain the propellant co-orbiting with Space Station Freedom until just before the mission launch date. At that time, the LTV would depart Freedom with the aid of an OMV, rendezvous with the co-orbiting propellant tankers, and transfer the propellant from Freedom. The benefit of this option is the reduction of the impact to Freedom's microgravity environment by reducing the mass stored on Freedom. Unfortunately, this method will require additional OMVs to control the propellant tankers, and it will add some operational complexity by having to rendezvous and transfer propellant from Freedom.

4.2.3.5 Meteoroid and Orbital Debris Shielding for Hangars

The protection of transfer vehicles from meteoroids and orbital debris while at Space Station Freedom involves balancing the benefits of additional protection against the costs of adding significant mass to the hangar walls. A hangar wall thickness was selected to provide some additional protection to EVA astronauts while operating in the unpressurized hangar. The selected meteoroid/debris shielding includes a main wall of 0.8 mm of Al alloy, an exterior bumper wall of 0.2 mm at a distance of 3 cm, and the gap between the walls partially filled with layers of multilayer insulation (MLI). The effectiveness of shields of various thicknesses is illustrated in figure 4.2.3-7. The figure compares the thin shield of the Explorer 46 satellite, able to stop microgram

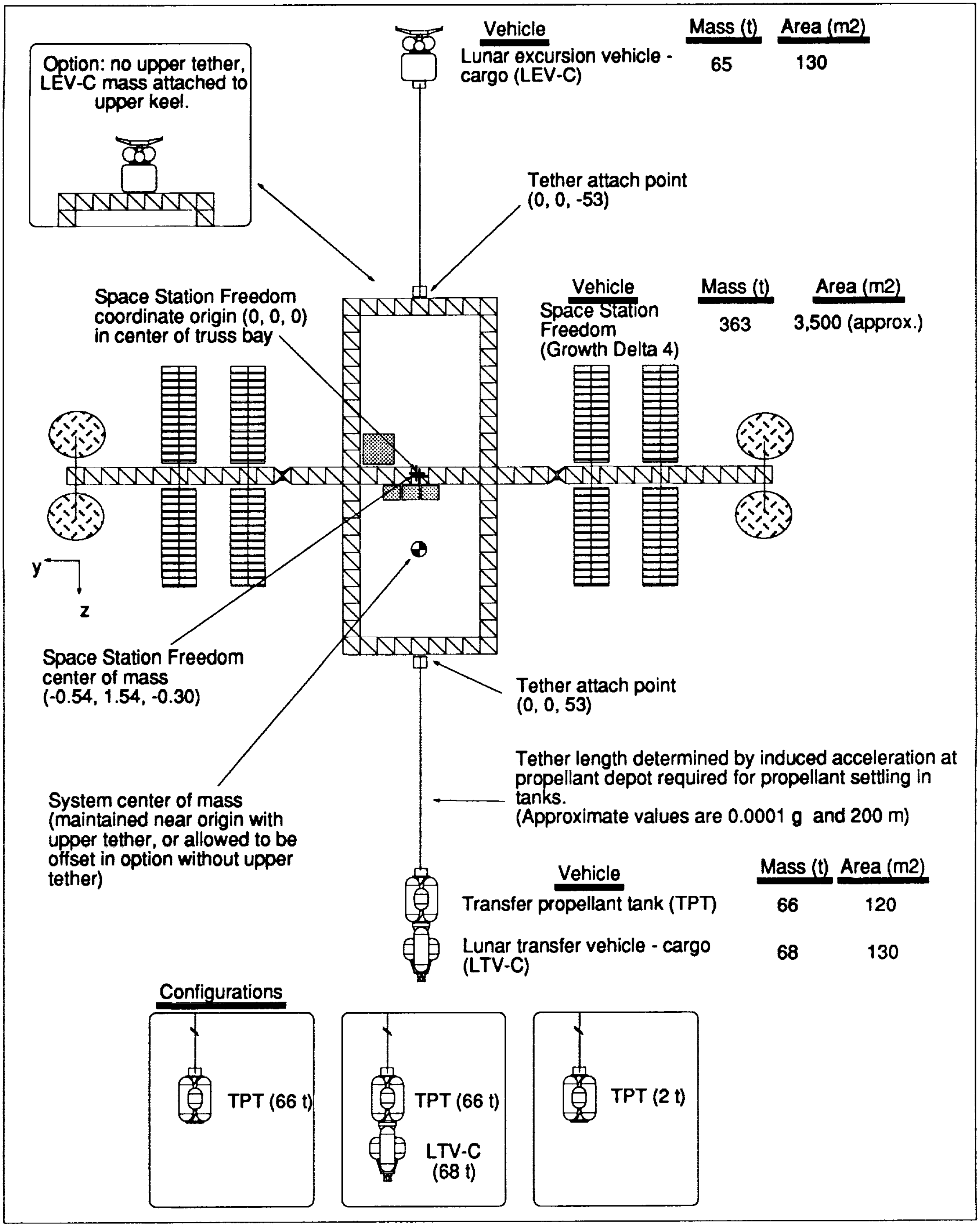


Figure 4.2.3-6.- Lunar Evolution Space Station Freedom with tethered propellant depot - flight 1.

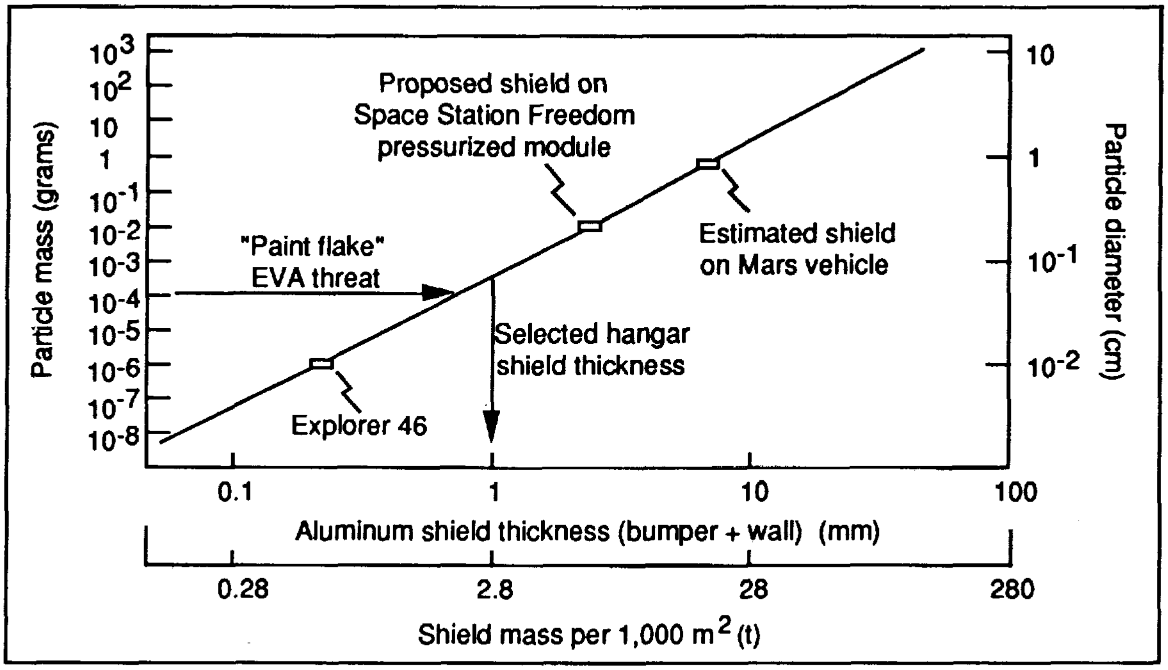


Figure 4.2.3-7.- Selection of 1 mm aluminum shield for vehicle hangar wall.

size particles, to the heavier shields for Freedom modules. The 1 mm shield selected would stop particles just under 1 mm in diameter and reduce the threat to EVA astronauts by roughly a factor of 10. This would remove the debris risk as an operational limit to EVA time in the hangar. The resulting mass penalty for the hangar walls is 2.8 t per 1,000 square meters of hangar wall.

Natural meteoroids include low-density cometary material in periodic meteor showers, and higher density asteroidal material. Meteoroids with high velocities 10 - 60 km/s) will arrive at Freedom with directions randomly distributed across the hemisphere above the horizon.

Artificial orbital debris consists of large objects that are tracked optically and with radar, and of the more numerous small objects and particles. Limited data are available on the number of small objects and particles. For modeling purposes, distributions are extrapolated from the numbers of larger objects and have uncertainties of a factor of 10 for the current environment. In the coming decades, the amount of debris is projected to increase by a factor of 2 to 10, depending on future space activity and debris countermeasures. Orbital debris impact directions are concentrated in the x-y (horizontal) plane. Typical impact velocities of orbital debris (10 km/s), although lower than meteoroids, are more likely to be capable of penetrating shields since particle fragments may survive the impact with the bumper shield. The retrieval of the Long Duration Exposure Facility satellite will significantly add to the understanding of the current environment.

The current space suit can be penetrated by particles as small as 0.5 mm diameter at orbital velocities. The probability of such an impact on an EVA astronaut in the current environment is roughly $2x10^{-6}$ per hour of EVA time. Without additional protection, the Space Station Freedom risk limit would require an operational limit of 3 to 5 EVA days per month for each astronaut.

4.2.3.6 Lunar Transfer Vehicle Accommodation

In addition to micrometeoroid protection, a hangar also provides a suitable operational environment for EVA operations and IVA teleoperated vehicle processing. The hangar is equipped with sufficient lighting, robotic access, orbital support equipment, and power to perform all processing activities. A station arm is mounted on a mobile track assembly like that shown in figure 4.2.3-8. This arm will have the capability of reaching all the LTV's orbital replacement unit mounts for ease of teleoperated vehicle servicing.

4.2.3.7 Lunar Transfer Vehicle Servicing and Refurbishment

Recent analysis indicates that a vehicle processing crew of four will require approximately 123.5 shifts (8-hour work periods) to completely process a reusable LTV on-orbit, assuming current level vehicle system designs. Figure 4.2.3-9 shows the required LTV processing activities for three major operational periods and provides a top-level breakdown of the required number of shifts to perform the various tasks. A complete discussion of LTV processing activities, including crew skill mix, re-

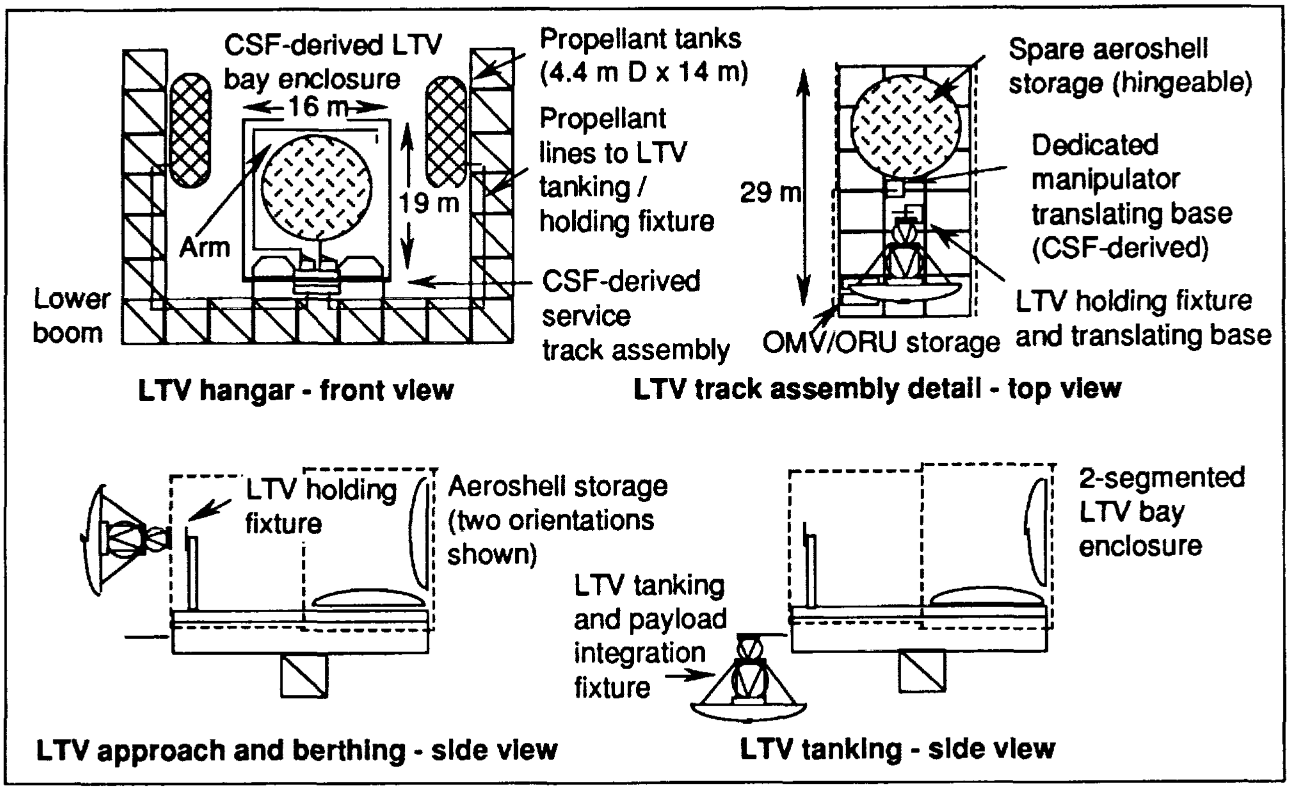


Figure 4.2.3-8.- Lunar transfer vehicle processing facility option.

source requirements, and task flows with traceability to current KSC processing activities is provided in Volume IV of the Exploration Studies Technical Report.

4.2.3.8 Issues and Summary

The major open issues that will impact Space Station Freedom accommodation of the Lunar Evolution case study are (1) the optimum on-orbit propellant storage technique, (2) a further definition of an acceptable micrometeoroid/orbital debris protection level, and (3) a refinement of the LTV processing requirements at Space Station Freedom.

4.2.4 Mars Evolution Case Study

4.2.4.1 Case Study Requirements

The Space Station Freedom mission requirements as stated in the SRD for the Mars Evolution case study are presented in table 4.2.4-I. The Mars transfer vehicle is not assembled on Space Station Freedom, but is accommodated on a free-flying, co-orbiting assembly fixture. Freedom's main requirements are to house the transient mission crew and Mars transfer vehicle (MTV) assembly crew, support the advanced development of Mars outpost and MTV systems, and provide a life sciences research capability.

4.2.4.2 Space Station Freedom Growth

The Space Station Freedom evolution growth deltas and the programmatic schedule required to support the mission requirements are provided in table 4.2.4-II and figure 4.2.4-1 respectively. Figure 4.2.4-2 shows the number of launches required to deliver Space Station Freedom's hardware growth elements to LEO. It is highly unlikely that Freedom will be able to evolve to delta 8 in time to support a 2005 launch date if growth assembly is limited to a Space Shuttle ETO capability. In addition, further definition of the required life sciences research program content and duration is required so that appropriate research accommodation facilities can be provided by OSS.

Table 4.2.4-III shows the currently understood Mars Evolution resource requirements to conduct life sciences and technology development research and assemble the MTV.

4.2.4.3 Growth Configuration Analysis

Based on the same analysis discussed in sections 4.2.3.3 and 4.2.3.4, the growth configuration shown in figure 4.2.4-3 is recommended to meet the Mars Evolution case study requirements. The configuration is gravity gradient stabilized, minimizing control requirements while

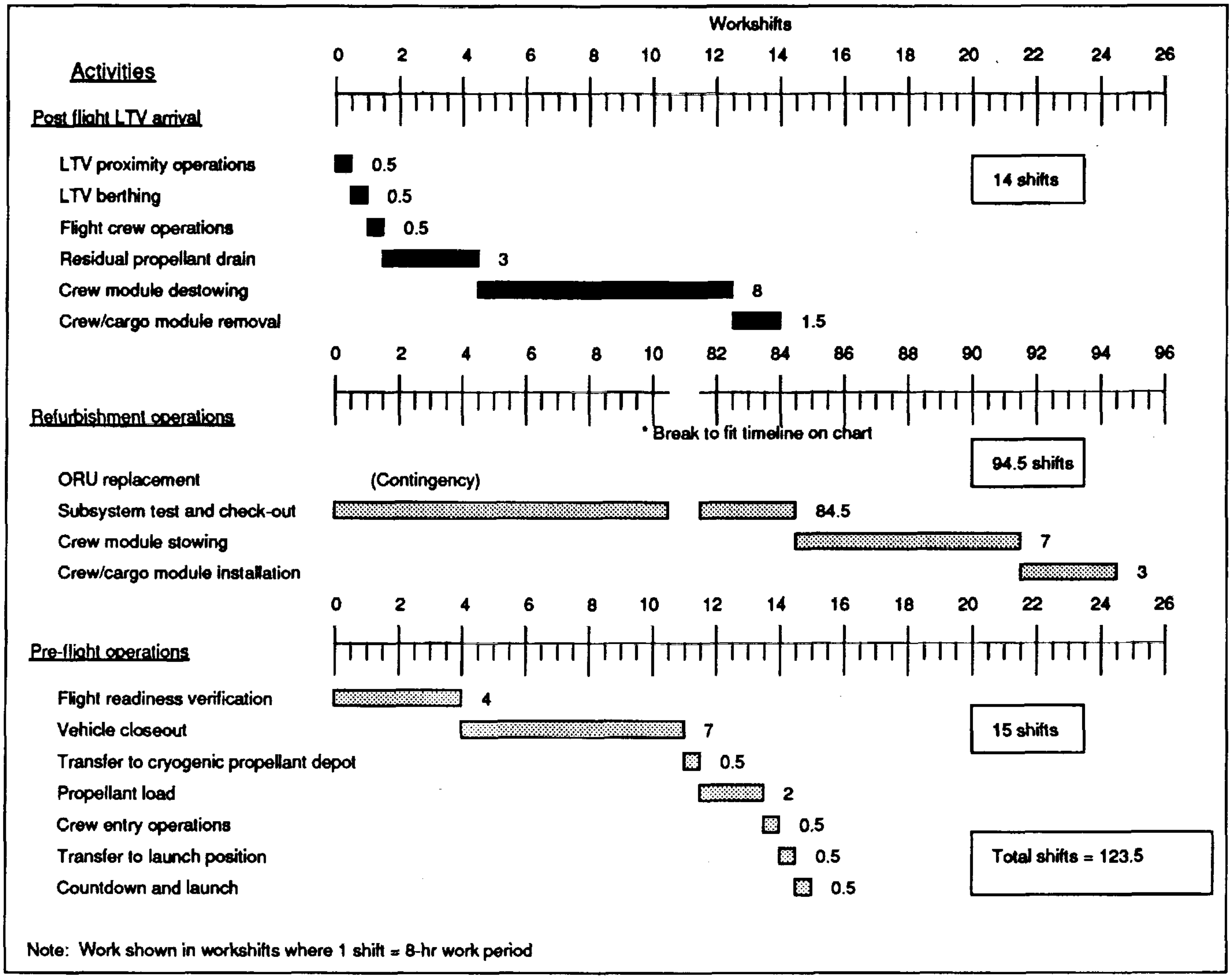


Figure 4.2.3-9.- Lunar vehicle processing operations.

TABLE 4.2.4-I.- MARS EVOLUTION CASE STUDY - SPACE STATION FREEDOM FUNCTIONAL REQUIREMENTS

- Provide capability to support advanced development systems for Mars outpost and space transportation.

- Provide capability for housing transient mission crew, support crew, and mission equipment.

-Transient mission crew: (Outbound)	Up to 2016, support mission crew of 5 Beginning in 2016, support mission crew of 7
-Mission support crew:	Accommodate support crew of 6, for up to 15 months, with 6-month tours of duty
-Mission equipment:	Beginning in 2005 accommodate 10 metric tons

- Provide isolation of outbound mission crew and quarantine of Mars crew and samples returning from first two missions.

TABLE 4.2.4-II.- MARS EVOLUTION CASE STUDY - SPACE STATION FREEDOM GROWTH DELTAS

Δ1	Two 25-kW solar dynamic modules; two 25-meter transverse boom extensions; space-based OMV and space-based OMV accommodations
Δ2	Upper/lower keels and booms
Δ3	One habitat module; two resource nodes
Δ4	Two 25-kW solar dynamic modules; servicing facility phase 1
Δ5	One large pocket laboratory (artificial-g); one large pocket laboratory (CELSS); servicing facility phase 2
Δ6	Life sciences laboratory module; two resource nodes
Δ7	Phase 3 servicing facility (completed CSF)
Δ8	One large pocket laboratory (quarantine facility)

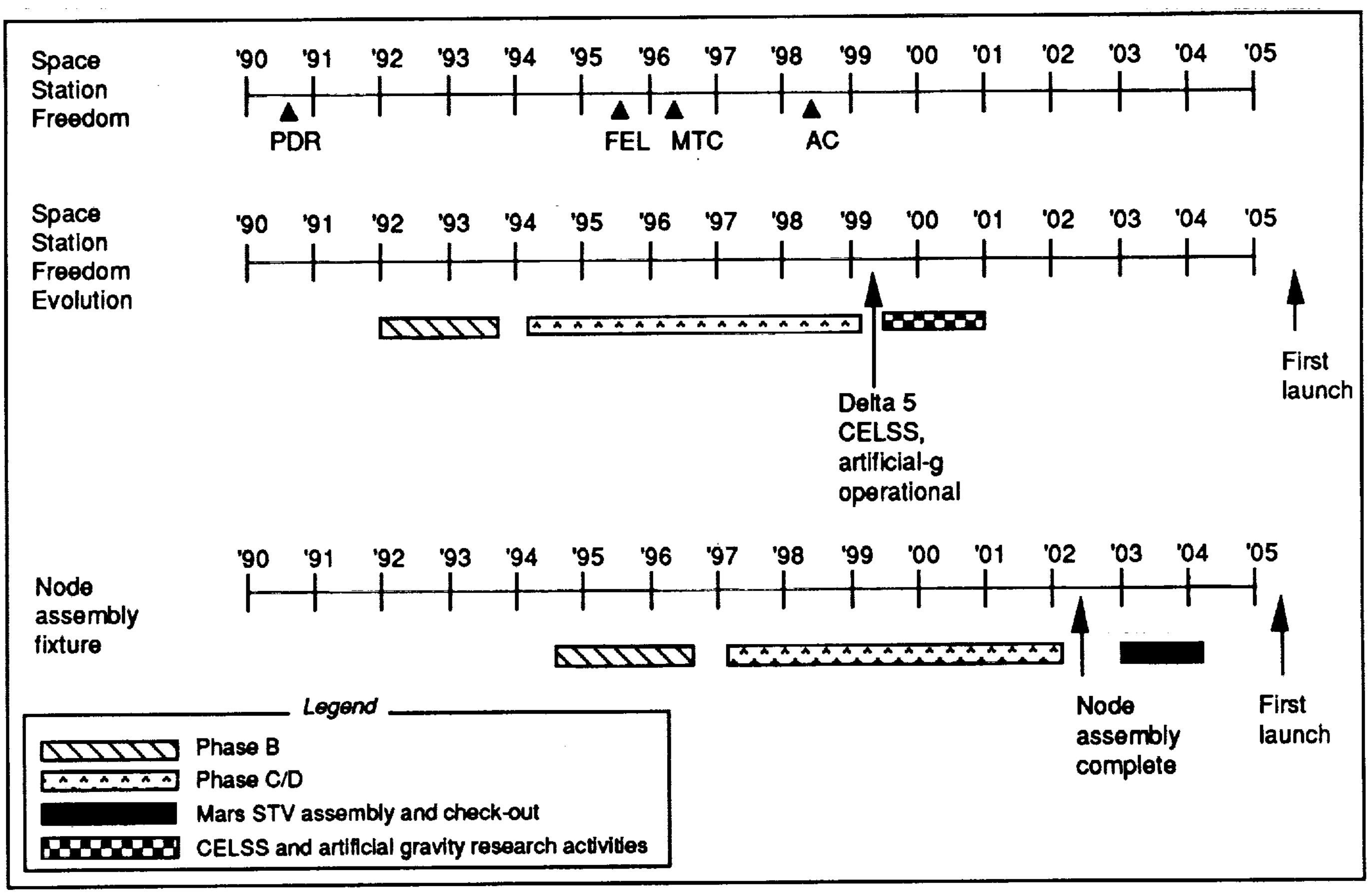


Figure 4.2.4-1.- Mars Evolution case study: programmatic schedule for Space Station Freedom evolution.

Sequence \ Flight		1	2	3	4	5	6	7	8	9	10	11
Mars Evolution manifest using Space Shuttle	Mass	●	●	●	●	●	●	●	●	●	●	○
	Volume	●	●	●	○	●	●	●	●	○	●	○
Mars Evolution manifest using Shuttle-C	Mass	●	○	○	○	○	●	○				
	Volume	●	●	●	●	●	●	●				

Legend

● - Greater than 90% of available launch vehicle mass/volume utilized for cargo delivery

○ - Less than 90% of available launch vehicle mass/volume utilized for cargo delivery

Figure 4.2.4-2.- Earth-to-orbit manifest options for Space Station Freedom Mars Evolution case study growth hardware.

TABLE 4.2.4-III.- MARS EVOLUTION CASE STUDY - SPACE STATION FREEDOM RESOURCE REQUIREMENTS

Station resource	Current recommendation
Power	
Average	175 kW
Peak	NA
Crew	18
Pressurized Modules	
US habitation	2
US laboratory	2
ESA laboratory	1
JEM laboratory	1
Pocket laboratory	3
Resource nodes	8
Transverse boom	
Dual keel	Scar for distributed systems
Servicing facility	Scar for facility and distributed systems
Power modules	Scar for addition of power modules at boom ends
Payloads	
APAE	18
Tether payloads	2

providing the appropriate facilities to conduct life sciences and technology research. In addition, other potential Space Station Freedom users, such as microgravity research, astrophysics, and Earth sensing communities would still have a suitable environment in which to operate. Table 4.2.4-IV lists the Freedom Station growth elements and their masses required to reach delta 8.

4.2.4.4 Mars Transfer Vehicle Departure Analysis and Implications

The MTV departure window durations from Space Station Freedom orbit are widely varying and require further investigation for each specific mission for an exact determination and optimization.

4.2.4.5 Issues and Summary

Although the SRD specified the use of a free-flying assembly platform, a trade must be conducted to assess the optimum location for assembling a Mars-class transfer vehicle. Although Space Station Freedom could probably accommodate the assembly of the MTV, it would most likely come at the expense of other Freedom users. Additional study is required to quantify these impacts to all Space Station Freedom users versus the cost of providing a separate assembly platform.

4.2.5 Mars Expedition Case Study

The Mars Expedition case study requirements of Space Station Freedom were limited to providing the capability to qualify life support systems for long duration flights, and access to Freedom's existing capabilities, provided the Mars Expedition requirements are com-

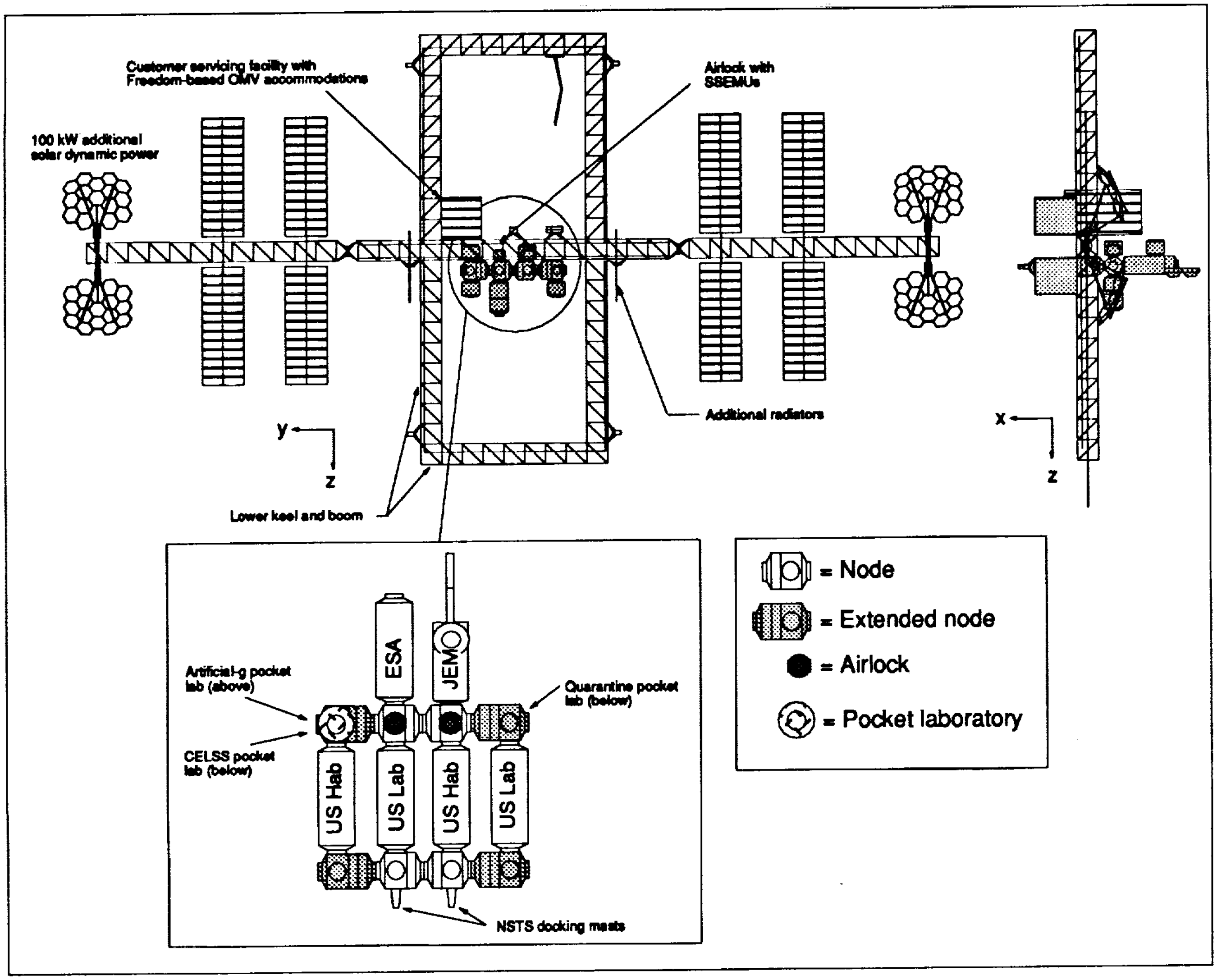


Figure 4.2.4-3.- Mars Evolution Space Station Freedom growth configuration (Delta 8).

TABLE 4.2.4-IV.- ADDITIONAL SPACE STATION FREEDOM MARS EVOLUTION ELEMENTS

Components	Number	Mass (kg)
Habitat	1	23,400
Life sciences laboratory module	1	34,700
Resource node	4	31,200
CELSS pocket laboratory	1	10,250
Artificial gravity pocket	1	10,250
Quarantine pocket laboratory	1	10,250
Truss bays	64	5,700
Utility trays	64	11,300
Solar dynamic units	4	27,600
Servicing facility	1	11,500
Attached payload accommodation equipment	4	1,450
Thermal radiators	2	6,300
Docking mast	1	1,400

patible with Space Station Freedom accommodation and utilization plans and commitments.

4.2.6 Options and Trades

The following section provides an overview of three important studies presently being conducted by OSS that will have a major impact on Freedom's ability to serve as a transportation node for human exploration.

4.2.6.1 Dynamic Analysis of Space Station Freedom-Based Mars Vehicle

The objective of this study is to define the expected low frequency dynamic characteristics of an evolutionary Space Station Freedom configuration used for Mars transfer vehicle assembly and to develop a reboost scenario for the flexible Space Station Freedom under active attitude control. A complete description of the NASTRAN (NASA Structural Analysis) finite-element model modal analysis and reboost scheme is presented in Volume IV.

4.2.6.2 Space Station Freedom Logistics Evolution

Currently, Space Station Freedom requires approximately five Space Shuttle flights per year to meet logistic resupply requirements. As a transportation node, the logistics requirements to support LTV and MTV processing activities and life sciences research and technology development requirements, as well as the additional consumables to support an increased crew, will dramatically increase the required number of Space Shuttle logistics flights. This study will quantify these requirements and assess the capability of the present logistics system to evolve to meet these increased resupply requirements.

4.2.6.3 Space Station Freedom Fluids Evolution

In order to support the currently defined exploration missions, Space Station Freedom will be required to store, handle, and provide thermal/fluid management of various fluids, the majority of which will be cryogenic propellants. The primary purpose of this study is to define fluid storage and handling strategies/requirements for the various case studies and their associated design impacts on Space Station Freedom. The four specific study tasks are: (1) conduct an inventory of all fluids expected to be associated with Space Station Freedom during its initial and evolutionary phases, (2) identify fluid management requirements such as storage, supply, transfer, handling, thermal, and safety issues, (3) develop optimal fluid management strategies and concepts for fluids accommodation to minimize scarring of Space Station Freedom and its operation, and (4) identify impacts to Space Station Freedom design and operation systems and subsystems identified in task 3 and determine the necessary hooks and scars to be included in Space Station Freedom Phase I design to allow future fluid requirements. A final report on tasks 1 through 4 has been completed and released by Lewis Research Center as a separate NASA contractor report (#185137).

4.2.7 Summary

The Space Station Freedom program has assessed the accommodation requirements and provided an appropriate implementation of those requirements for each of the FY 1989 case studies. This implementation included an engineering description of the systems utilized to meet the mission requirements, including habitat modules, laboratory modules, and vehicle processing facilities. Preliminary analysis shows that the Space Station Freedom program can support all Lunar Evolution case study mission requirements by evolving the current Freedom configuration into a transportation node, provided all current Space Station Freedom hooks and scars are maintained in the baseline program. Potential problems may arise in the ability of Freedom to evolve to provide an orbital facility for research into extended crew stay times in order to support the Mars Evolution and Mars Expedition case study launch dates.

4.3 TELECOMMUNICATIONS, NAVIGATION, AND INFORMATION MANAGEMENT

This section describes the telecommunications, navigation, and information management (TNIM) architectural and system design functions for the lunar and Mars case studies, and identifies critical space and Earth-based technology that must be developed prior to initiation of the mission implementation phases.

4.3.1 Lunar Evolution Case Study

The 1989 study effort emphasized the telecommunications system, deferring detailed study of navigation and information management to future years. This subsection discusses the TNIM needs of the lunar evolution mission elements and describes a reference-point tracking and data acquisition (T&DA) architecture capable of providing telecommunications and tracking support.

4.3.1.1 Reference Point Architecture

A reference model of elements to establish the services necessary for lunar exploration was defined. Lunar element TNIM needs were derived from an analysis of the reference model. The major drivers of the T&DA system are the number of links required to serve the users and the data return rates on those links. Derived-link and data-traffic needs were used to conceptually design a T&DA architecture. The link requirements estimated

for transportation vehicles and surface elements are illustrated in figure 4.3.1-1. A candidate point design for a T&DA system that serves as a reference point architecture is illustrated in figure 4.3.1-2.

Depending on user location, communications and navigational needs of lunar users would be supported in different ways by the T&DA system. Communications radio coverage requirements are met for users within a 10-km radius of a nearside lunar surface terminal by interfaces and a broadband radio frequency wide area network, which allow flexibility in the deployment of both users and the surface terminal. Primary and backup antennas and 30-meter tall towers are provided to support the UHF wideband local area communications system. A capability exists to switch the backup system antenna should the primary antenna become inoperative or if the primary tower should be damaged. The backup tower and antenna are located a minimum distance of 1 km from the primary system. On the farside,

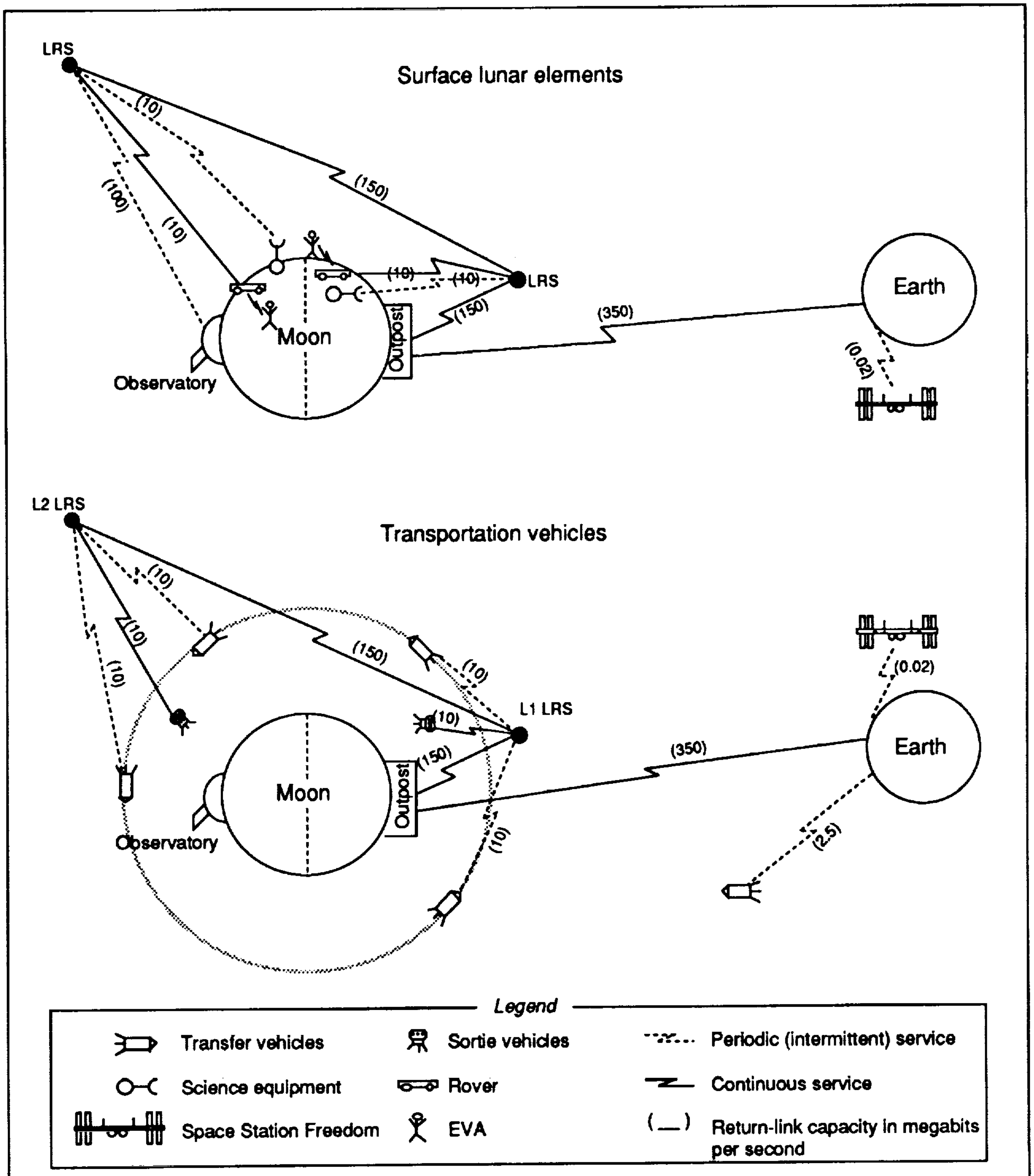


Figure 4.3.1-1.- Return links to satisfy lunar elements.

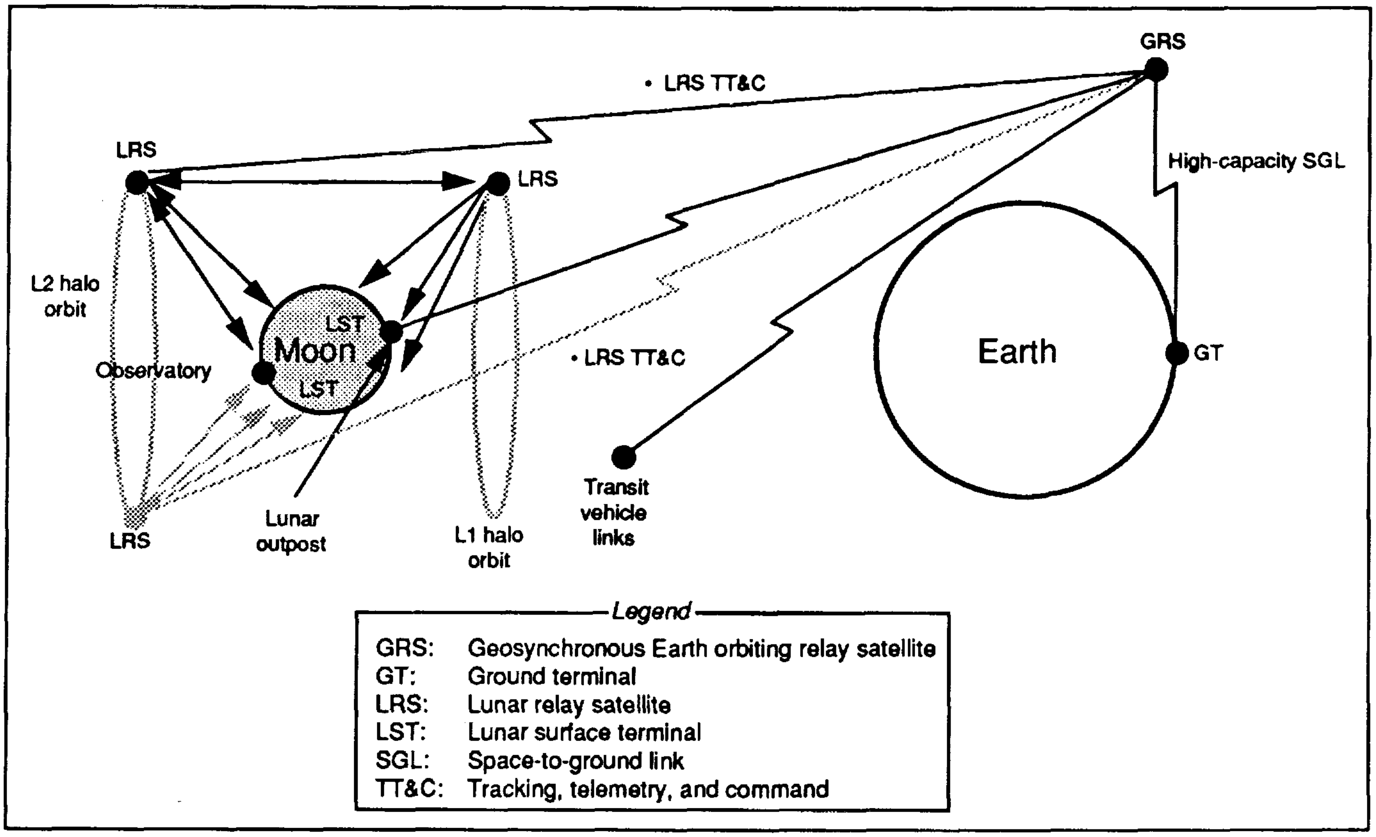


Figure 4.3.1-2.- Lunar T&DA system reference architecture.

the instruments of the very-low frequency array and other users within 10 kilometers of a surface terminal would be connected by hardwire links, either coaxial or fiber-optic cable. Users more than 10 kilometers from the surface terminals would communicate through radio frequency links with lunar relay satellites, as would users in LLO or in lunar ascent or descent. Users in transit between LEO and LLO would be similarly supported directly by one geosynchronous relay satellite.

The geosynchronous relay satellite, in an inclined orbit, would provide the primary relay for all communications between Earth and the Moon by linking a United States-based ground terminal with the nearside surface terminal located at the lunar outpost. Nearside lunar elements would be supported by the surface terminal and a lunar relay satellite in halo orbit about the L1 libration point. Farside elements would be similarly supported by a terminal collocated with the farside observatory and a relay satellite in halo orbit about the L2 libration point. The significant characteristics of this reference architecture are listed in table 4.3.1-I.

To reduce operational complexity and risk, geosynchronous and lunar relay satellites would support all communications by frequency-translated (bent-pipe) links with no demodulation or other processing performed onboard. Terminals on the lunar surface and on the ground would act as data terminals, demodulating and modulating communications signals and providing distribution functions. In addition, the nearside terminal would act as a major processing center, supporting the nearside outpost as the center for lunar operations.

Some users of the T&DA system will need to provide various levels of data processing (including compression, storage, and buffering) to remain within the coverage and capacity limits imposed by the system. For example, data from scientific instruments on the lunar surface will need to be buffered until a T&DA contact can be scheduled for data communications.

Capacity limitations and conflicts among users may further require that some elements be denied service during particular periods. For example, when three transport vehicles are in LLO and simultaneous crew surface activities require support, service to surface instruments may be delayed.

The following paragraphs describe in more detail the elements of the system to support the Lunar Evolution case study.

Ground Terminal. A single ground terminal located within the continental United States would provide the Earth interface for the T&DA system. Return lunar com-

TABLE 4.3.1-I.- CHARACTERISTICS OF EARTH-MOON REFERENCE COMMUNICATIONS ARCHITECTURE

System element	Supporting subsystem	Application
Ground terminal	1 GT-GRS antenna - 20-meter diameter - Ka-band (20.7 GHz)	Supports link to GRS
Geosynchronous relay satellite	1 GT-GRS antenna - 2-meter diameter - Ka-band (20.7 GHz)	Supports link to GT
	2 lunar service antennas - 4-meter diameter - Ka-band (32.5 GHz)	One antenna supports link to nearside LST One antenna supports link to single transit user
	2 farside LRS TT&C antennas - 1-meter diameter - Ka-band (32.5 GHz)	Supports TT&C links to farside LRSs
Nearside lunar surface terminal	1 LST-GT antenna - 4-meter diameter - Ka-band (32.5 GHz)	Supports Moon-Earth communications link to GRS
	1 LST-LRS antenna - 3-meter diameter - Ka-band (32.5 GHz)	Supports link to nearside LRS and thereby farside users, the farside LST, and nearside users beyond 10 km of nearside LST
	Wide area network - UHF (400 to 800 MHz) - 30-meter antenna mast provides 10-km line-of-sight visibility	Supports lunar users within 10 km of nearside LST
	Hardwire interfaces (fiber-optic or coaxial cable)	Supports high data rate users within 10 km of nearside LST
Nearside lunar relay satellite	1 multibeam antenna - 3-meter effective aperture - 5 beams - Ka-band (32.5 GHz)	4 beams support nearside users 1 beam supports forward/return link to nearside LST
	1 crosslink antenna - 3-meter diameter - Ka-band (30 GHz)	Supports relay to farside LRS
Farside lunar relay satellite	1 multibeam antenna - 3-meter effective aperture - 5 beams - Ka-band (32.5 GHz)	4 beams support farside users 1 beam supports farside LST
	1 crosslink antenna - 3-meter diameter - Ka-band (30 GHz)	Supports relay to nearside LRS
	1 TT&C antenna - 1-meter diameter - Ka-band (32.5 GHz)	Supports TT&C to/from GRS and contingency Earth-farside communications
Farside lunar surface terminal	1 farside LRS antenna - 1.5-meter diameter - Ka-band (32.5 GHz)	Supports link to farside LRS
	Hardwire interfaces (fiber-optic or coaxial cable)	Supports farside users within 10 km of farside LST

munications relayed by the geosynchronous relay satellite would be demodulated and distributed to Earth-based users by the ground terminal. Forward communications to lunar elements would be multiplexed, formatted, modulated, routed, and transmitted. The ground terminal performs range and range-rate extraction of one-way and/or two-way return transit signals from the element to provide tracking data. The ground terminal would support the other T&DA system elements, providing tracking, telemetry, and command (TT&C) links, as required, to control other T&DA elements.

All primary T&DA system links would use the Ka-band to permit contingency ground terminal support. The primary ground link to the geosynchronous relay satellite would be supported by a 20-meter Ka-band (20.7-GHz) antenna capable of maintaining an end-to-end bit error rate of 10^{-6} in the presence of interference due to rain.

<u>Geosynchronous Relay Satellite</u>. A single satellite would support a continuous link to the ground terminal, links to the nearside lunar surface terminal and lunar elements in transit between LEO and LLO, and low-data-rate TT&C links to the farside lunar relay satellites. A single 2-meter Ka-band (20.7 GHz) gimballed antenna would support the bent-pipe relay of return and forward channels to and from the ground terminal. These channels would contain the frequency-division multiplexed 350-Mbps return link from the lunar surface terminal, the multiplexed 90-Mbps forward link to the lunar surface links to lunar elements in transit, TT&C, and geosynchronous relay satellite data.

The links to transiting lunar elements and the nearside terminal would be supported by two 4-meter Ka-band (32.5 GHz) gimballed antennas, each providing a single beam. The forward and return links to transit users would each provide 20-MHz bandwidth to support navigation and data transmission up to 10 Mbps. The nearside forward link would support a 90-Mbps frequency-division multiplexed data stream to the Moon, and the return link would support a 350-Mbps frequency-division mulitplexed data stream back to Earth.

A single geosynchronous relay satellite, even with an inclined orbit, would not maintain continuous visibility with the nearside lunar relay satellite. However, only relatively short service gaps are estimated to occur on several days each month. A second relay satellite in geosynchronous orbit would be needed if continuous coverage is required.

Finally, the satellite would need two 1-meter Ka-band (32.5 GHz) gimballed antennas to support low-data-rate TT&C to the farside lunar relay satellites. In the event of disruption of the crosslink between the farside and nearside satellites, the link between the farside relay satellite and the geosynchronous relay satellite may be used to maintain farside service at reduced data rates.

Nearside Lunar Surface Terminal. The nearside lunar surface terminal would be collocated with the nearside outpost, directly servicing the surface habitats, scientific facility, mining facility, and nearside observatory. Moreover, the terminal would interact with the nearside relay satellite to support both nearside and farside elements with local communications routed through the terminal. Thus, the terminal would serve as a communications center for all activities, in addition to providing the primary Moon-to-Earth communications link. Forward communications from Earth would be demultiplexed and distributed to lunar elements, and return communications from the Moon would be multiplexed and formatted for relay to the geosynchronous relay satellite.

A 3-meter Ka-band (32.5 GHz) antenna would support forward and return communications with the nearside lunar relay satellite; the link to the satellite would be the frequency-division multiplexed (FDM) composite of nearside and farside channels. Return communications used by the nearside outpost (e.g., long-distance rover video) would be distributed locally by the nearside surface terminal using the UHF network. Return communications destined for Earth would be demodulated and multiplexed with other data and remodulated for trunk transmission.

The primary 350-Mbps return and 90-Mbps forward frequency-division multiplexed links to the geosynchronous relay satellite would be supported by a 4-meter Ka-band (32.5 GHz) antenna at the nearside surface terminal. All forward communications from Earth would be demodulated and demultiplexed to allow routing and distribution to elements at the nearside base or remodulation and relay to the lunar relay satellite(s).

Nearside Lunar Relay Satellite. A single satellite in halo orbit about the L1 libration point would provide nearside element communications coverage and a crosslink to the farside relay satellite. The halo orbit would have a period of 2 weeks and would provide nearside lunar element visibility. An average of 70 percent or more of the lunar hemisphere would be visible with coverage gaps near the lunar limbs and poles.

A Ka-band gimballed multibeam antenna with an effective aperture of 3 meters would support nearside communications between nearside elements and the nearside terminal. Different frequency assignments would enable concurrent support of lunar elements in close proximity. Elements would require transceivers with multiple frequency capability; otherwise, they would be subject to operational constraints.

The four beams represent a shared communications resource to support as many as three concurrent lunar transport vehicles, one long-distance rover, one scientific orbiter, and as many as 16 dispersed scientific instruments. Techniques such as scheduling service, polling unmanned instruments, or setting demand-access protocols would be used to establish communications and navigational links.

The fifth beam of the L1 multibeam antenna would support the relay to and from the nearside lunar surface terminal, including the L1 satellite TT&C. Return farside and nearside links would be frequency translated and relayed as FDM downlinks to the nearside terminal. Forward farside and nearside user links would be frequency translated and switched to the crosslink antenna or the appropriate beam of the Ka-band multibeam antenna.

A 3-meter Ka-band (30-GHz) gimballed crosslink antenna would adequately support the bent-pipe relay of links to/from the L2 satellite. Farside forward channels uplinked from the nearside terminal would be frequency translated and transmitted as a 105-MHz bandwidth FDM crosslink to the farside relay satellite. Farside 200-MHz bandwidth FDM return crosslink channels would be frequency translated and switched to the nearside terminal FDM beam of the Ka-band multibeam antenna.

Farside Lunar Relay Satellite. A single satellite, similar to the one at L1, would be placed in a halo orbit about L2. This satellite would support farside coverage and provide a crosslink to the L1 satellite. A second L2 satellite is explicitly indicated in figure 4.3.1-2 (referenced earlier) to emphasize the advisability of on-orbit redundancy to maintain farside service if the primary satellite fails. This second satellite would also provide more complete coverage of the limbs and poles and visibility to more than 70 percent of the farside hemisphere. However, only one operational L2 satellite would be required at a given time to satisfy currently identified farside service needs. Both L2 satellites would maintain low-data-rate TT&C links with the geosynchronous re-

lay satellite.

A Ka-band gimballed multibeam antenna with an effective aperture of 3 meters would support farside lunar communications, and a 3-meter Ka-band gimballed crosslink antenna would support the bent-pipe relay to and from the L1 satellite. Four high-gain spot beams would each support 20-MHz forward and return communications and navigational links for farside elements; the fifth would support forward and return communications with the farside lunar surface terminal. However, unlike the L1 relay satellite, all multibeam return communications would be frequency translated and switched to the crosslink antenna for relay. Conversely, all multibeam forward communications would be received as an FDM signal through the crosslink antenna, frequency translated, and routed to the appropriate beam.

To ensure control and monitoring of the L2 satellites in the event of crosslink failure, a 1-meter gimballed Ka-band antenna would maintain the L2 satellite TT&C link with the geosynchronous relay satellite. As a contingency, this antenna might be used to relay farside communications to the geosynchronous satellite at reduced data rates.

Farside Lunar Surface Terminal. The farside terminal would be simpler than the nearside one, and it would be collocated with the farside lunar observatory, directly servicing the VLF array and possibly other users through hardwire interfaces. Fiber-optic or coaxial cables would link lunar elements within a 10-kilometer radius of the farside terminal.

A 1.5-meter Ka-band antenna would support forward and return communications with the L2 satellite, transmitting a 100-Mbps time-division multiplexed return link and receiving less than a 1-Mbps time-division multiplexed forward link. Forward communications from the L2 satellite would be demodulated and distributed to the appropriate farside elements, and farside return communications would be multiplexed and formatted for transmission to the L2 satellite.

4.3.1.2 System Development Schedule

Development of the TNIM infrastructure required for lunar exploration is expected to be staged over time. The first phase will probably be a technology and flight readiness demonstration, commencing as soon as the exploration program is initiated. This phase would be completed prior to beginning the C/D development phase for relay satellites, the ground terminal, or lunar surface terminals. An initial schedule analysis, which presumes normal acquisition and delays, suggests that initial system availability could be as late as 2007.

4.3.1.3 Relationship to Other NASA Programs

LEO telecommunications support of the lunar exploration program could be provided by the Advanced Tracking and Data Relay Satellite System (ATDRSS), which is scheduled to begin operating in 1997. The ATDRSS is already planning to support the STS and Space Station Freedom. Supporting the lunar exploration program when it uses STS and Freedom should not require major modifications to the existing ATDRSS design. LEO T&DA support for orbital transfer vehicles (OTVs) could be provided if the communications systems onboard OTVs are compatible with the Ka-band systems chosen for ATDRSS.

Freedom could be involved in technical demonstrations of new technology required for the lunar T&DA system, such as Ka-band multibeam antennas.

The relay satellites and lunar surface elements of the T&DA system will require launch support for deployment. This requirement would include launch and deployment of the nearside lunar surface terminal, launch of a geosynchronous relay satellite, and launches of lunar relay satellites.

4.3.1.4 Options and Trades

The reference architecture described previously is one of several alternatives. Lunar communications and navigational studies are still in preliminary stages, and alternative T&DA architectures are still being assessed. The alternative architectures differ from the reference point architecture in at least one of the following ways:

a. Placement and/or use of lunar relay satellites

b. Use of ground terminals instead of a geosynchronous relay satellite

c. Variation in interfaces or attributes of the communications system elements

Requirements for the lunar relay satellites depend entirely on mission scenario and user need, with some conceivable scenarios requiring no lunar relay satellites. The mission options are as follows:

a. The relay satellites in lunar halo orbit about the L2 libration point enable farside visibility.

b. The relay satellites in halo orbit about the L1 libration point enable smaller latency delays from the lunar outpost for remote surface and LLO elements

located about the lunar nearside. They also enable smaller, less powerful stationary transponders for surface users.

c. Lunar surface coverage requirements at the limbs and poles and/or system reliability needs may lead to redundant satellites at each halo orbit.

d. Locating relay satellites at the L4 and L5 libration points might decrease the stationkeeping burden, compared to satellites at the L1 and L2 halo orbits. However, using those locations would increase latency delays and transponder requirements.

e. Another option is an architecture with relay satellites in LLO. However, this option would substantially complicate the relay system by requiring extensive use of crosslinks and handovers.

f. An architecture that does not use relay satellites is also being considered. This architecture would instead use fiber-optic or microwave trunks to interconnect the surface users. This architecture does not provide flexibility to the dispersed users compared to a satellite-based architecture, and might have some maintainability drawbacks, particularly in high-traffic areas.

Determining whether to use a geosynchronous relay satellite-based or a ground terminal-based architecture is a critical design choice that could greatly impact the availability, cost, and operational complexity of the overall lunar communications system. Preliminary analysis favors the relay satellite over the ground terminal, but further study is required to validate the result of this analysis.

Comparisons of the various architectural alternatives have begun. The alternative architectures will be evaluated and ranked according to the following general criteria: ability to satisfy user needs, system implementation and operations costs for a 20-year life cycle, and technological feasibility and risk.

Satisfying user needs is a major goal of the lunar T&DA architecture. This criterion will be measured by comparing attributes such as system availability, reliability, and data latency times. In addition, user impacts such as antenna size, pointing accuracy, number of handovers required, and power requirements will be compared. Considerations, such as the long-term reliability of tracking stations on foreign soil, and technical issues regarding the relative service provided by Earth stations and geosynchronous relay satellites will also be assessed. Future studies will develop and assess operations concepts and attempt to limit operational impacts to users.

Another major criterion is technological feasibility and risk, which will be measured by assessing the technology requirements for the alternative architectures, the current and projected availability of required technologies, and the complexity of the system element designs.

In addition to architectural trade studies, a study to determine the relative stability of the lunar halo orbits about the L1, L2, L4, and L5 libration points is currently in progress. This study is expected to determine if lunar relay satellites can be maintained in these orbits and the feasibility of station-keeping.

4.3.1.5 System Issues

Several TNIM issues regarding the Lunar Evolution case study remain; the most critical relate to data systems. Existing approaches to T&DA and mission control may not be acceptable for this program, because they depend heavily on full-time human console control of routine operations. Increased autonomy for lunar operations may be mandatory. Alternative distributed data system architectures that provide autonomous lunar operations with Earth backup must be developed and compared to alternative concepts.

Technology development will be needed for such system components as the lunar relay satellite Ka-band multibeam antenna; the nearside lunar surface terminal UHF broadband wide area network; data compression for video and scientific data; source coding techniques for data reliability and privacy; suitable relay satellite designs; and compact lightweight antennas, tracking systems, and transceivers for lunar elements.

Telecommunications requirements have been directly addressed in the reference architecture. Some requirements have significant potential impacts on the reference architecture. For example, the requirement that latency be minimized for real-time communications between the nearside outpost and the long-distance rover has led to the use of a nearside lunar relay satellite. The requirement that farside communications be maintained with the nearside outpost contributes to the need for an L1 satellite and cross communications between nearside and farside relay satellites. The need to restrict Earth facilities to the continental United States is one motivation for the use of geosynchronous relay satellites rather than globally distributed ground terminals. If continuous Earth-to-Moon communications are required without even short service gaps, more than just the reference architecture's single geosynchronous relay satellite may be required. Continuous communications for distributed lunar users, such as rovers located at the lunar poles and limbs, would further increase the required number of lunar relay satellites.

4.3.1.6 Conclusions

Continuing studies of lunar exploration will define the TNIM systems required to support it. Although studies must be completed, the reference point architecture provided appears to provide the foundation for the T&DA support required for any conceivable future exploration and use of the Moon. It appears that a feasible but complex system can be developed to provide the necessary telecommunications and navigational support.

Data system requirements must now be carefully assessed and data system architectures developed that can provide the operations control, telemetry processing, and autonomy required for affordable lunar exploration. These systems could have major impacts on the user service, life-cycle costs, and technological feasibility and risk of the TNIM system.

Analysis of alternative TNIM system concepts and architectures must be continued. Prior to NASA selection of a mission scenario recommendation, studies of lunar exploration TNIM systems should focus on (1) a detailed information management and data systems study, (2) development of operations concepts, and (3) trade studies that refine T&DA system designs and include operations and maintenance overhead.

4.3.2 Mars Evolution and Expedition Case Studies

Because the requirements are similar for both Mars case studies, all Mars-related information is combined in this section. The objective is to provide a high-performance TNIM system design that (1) meets and assists in establishing human-mission requirements while supporting general NASA exploration needs, (2) includes a functional TNIM architecture that incorporates low-complexity, manageable interfaces in both implementation and operational phases, (3) provides appropriate interface trade-space models to guide the development of an efficient, balanced end-to-end design, (4) evolves from current proven designs and accommodates performance upgrade options during the mission lifetime, (5) incorporates functional redundancy for all critical capability, and (6) identifies critical flight and ground technology development needed for subsequent implementation. An additional key objective is to support mission Integration Agents in developing related TNIM internal and interface system design.

The following sections outline the proposed partitioning plan for mission and TNIM functions, key architectural precepts, the preliminary architecture and expected system performance, critical technology needs, an implementation approach and schedule, conclusions and design issues, and FY 1990 study recommendations.

4.3.2.1 Partitioning of TNIM and Related Mission Support Functions

The primary criteria used in recommending the following mission and TNIM functional partitioning are to (1) engender the development of manageable and robust interfaces of relatively low complexity, and (2) aggregate like functions in efficient configurations that can support all NASA exploration activity at Mars.

OSO provides: (1) overall TNIM architecture, (2) dedicated Earth-based telecommunications and tracking stations, TNIM network control center, and information management system design using flight/ground distributed processors, (3) in situ Mars telecommunications relay satellite network, including modes for local mission control, (4) Earth-based radiometric and navigation support, which can use onboard optical/radiometric target observables, for initializing onboard real-time navigation functions (includes inter-spacecraft/habitat link acquisition predicts), and (5) a local reference navigation network using GPS-like navigation satellites about Mars and global surface radio beacons.

The Manned Flight Mission provides: (1) relay satellite delivery to Mars orbit, (2) all mission flight and planetary surface TNIM related subsystems compatible with the overall TNIM system design and architecture, (3) appropriate data compression and storage functions on mission element terminals, (4) onboard navigation and guidance for aerobraking, landing, docking, and other real-time navigation needs, and (5) mission operations and analysis areas, and all telemetry, command, and science data processing above level zero.

4.3.2.2 Mars TNIM Functional Design Precepts

The key functional design precepts assumed as guidelines for developing the TNIM design concepts are described below.

In keeping with the minimum cost objective of the expedition case, the TNIM network design will provide only the limited performance and operational lifetime (~5 years) necessary to support a single manned flight to Mars. In contrast, for the evolution case, the TNIM network will be a long-life (≥15 years), capital investment capable of supporting all manned and unmanned missions envisioned in an extended Mars exploration program.

The TNIM design will evolve from existing, proven capabilities with the incorporation of advanced technologies when justified by favorable cost/benefit trades. TNIM will be based on a balanced end-to-end system design to minimize the overall mission and TNIM network costs and maximize robustness. TNIM functions

will be combined in the network design where appropriate to reduce complexity and cost. Simple, manageable interfaces are very important to reduce costs and mission complexity, and to partition tasks in logical units. As an operational option to Earth-based control, local Mars TNIM control may be selected to facilitate real-time operations during human presence at Mars.

The TNIM design will provide emergency modes as well as redundancy to ensure the navigation and telecommunications systems are the last to fail.

4.3.2.3 **Mars TNIM System Design and Performance**

Telecommunications. The overall telecommunications design balances spacecraft and ground capability for the highest end-to-end performance and reliability as well as for lowest expected cost and complexity, using technology expected to be available in the 1996 to 1998 time period.

In situ Mars relay satellites are employed, with two Mars-stationary satellites each included in the baseline relay networks for the expedition and evolution cases. A relay satellite network provides significant improvements in connectivity and performance for both the Mars-Earth and the local Mars links. Figure 4.3.2-1 depicts a typical example of the dramatic improvement in connectivity achievable for the Mars Expedition. The connectivity and performance improvements provided by the relay network allow the use of simpler, lower performance communications subsystems for the mission elements.

The relay network also provides redundant and backup communications links as well as emergency links with greatly increased data rate performance and connectivity. In addition, a relay satellite in the Mars system can support a ranging link with an approaching spacecraft to provide important Mars-centered navigation range data beginning a few days before the critical aerocapture maneuver.

The characteristics and connectivity configuration of the key high-rate telecommunications links are summarized in figure 4.3.2-2 for both the expedition and evolution point designs. Antenna size, transmit/receive frequencies, data rates, and RF power are indicated for each link. Total dc power is also shown for each element. Ka-band is used for the high-rate space-to-Earth links and in situ links at Mars, X-band is used for the Earth-to-space links and direct emergency links to Earth, and UHF is used for local low-rate links at Mars. A single 34-m antenna is used to provide 20 Mbps uplink capability for either case; however, each additional uplink requires a separate antenna. Downlink data rates of 10 Mbps (relay satellite) and 20 Mbps minimum (Mars piloted vehicle) are provided using dual- or quad-arrayed 34-m antennas for the Expedition and Evolution case studies respectively. Local Mars system data rates up to 50 Mbps are available to support telerobotics and video monitoring. Higher peak data rate requirements are accommodated using large on-board buffer storage, time multiplexing, and multiple parallel vehicle-to-Earth links. The design also provides emergency capability using omni-directional antennas on the mission elements for X-band trans-

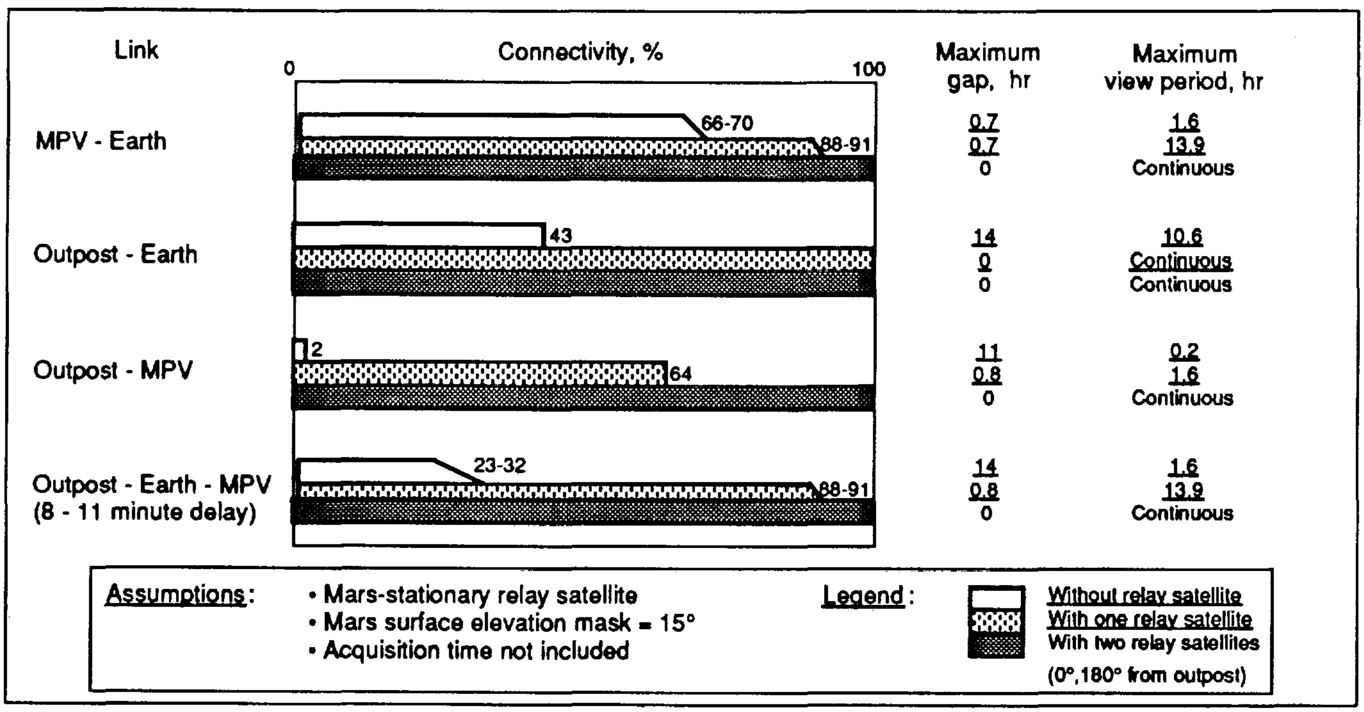


Figure 4.3.2-1.- Typical connectivity - Mars Expedition case (2002 option).

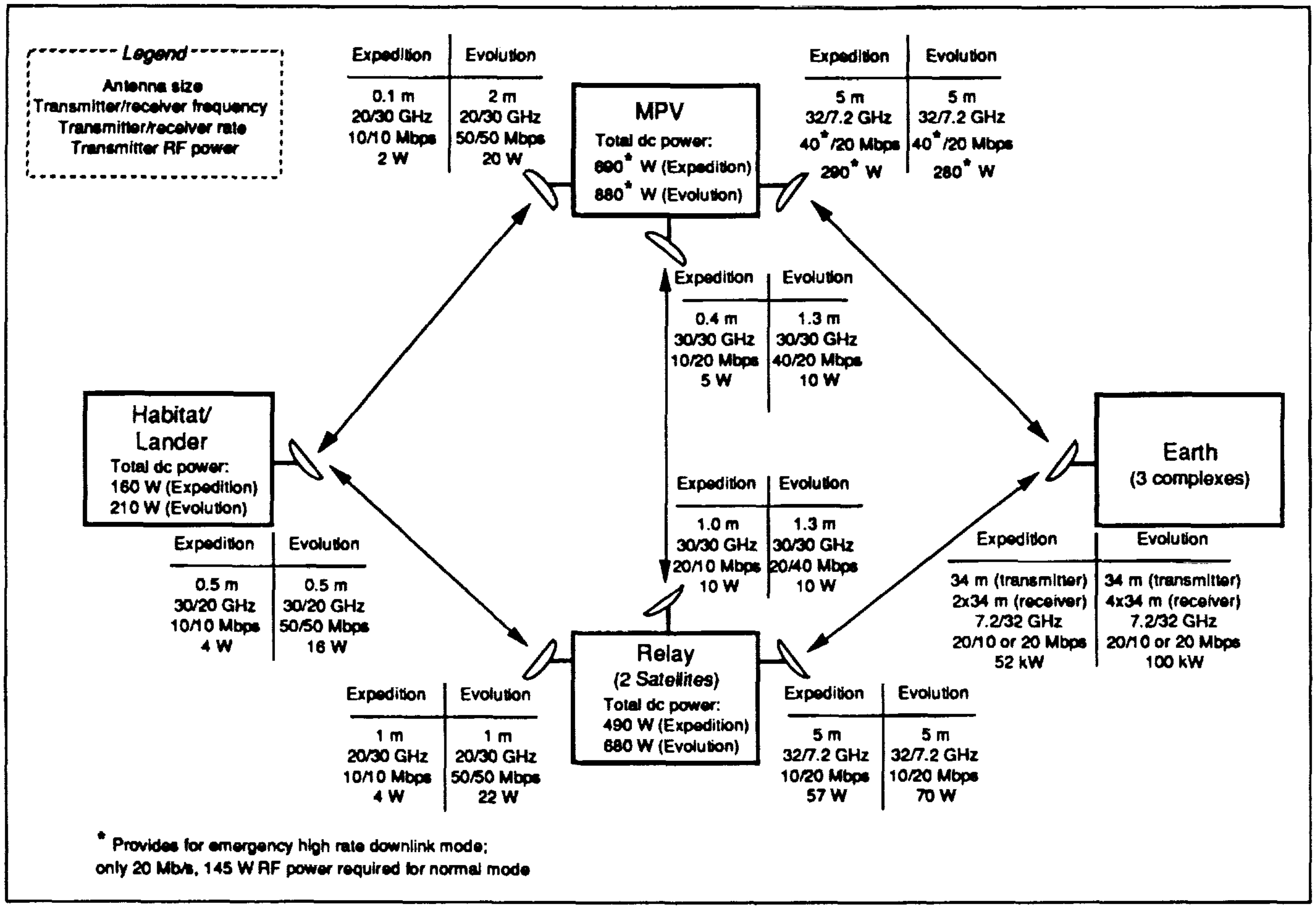


Figure 4.3.2-2.- Summary of point designs for key Mars links.

mission direct to Earth (<20 bps) or UHF/Ka-band via a Mars relay satellite (~20 kbps). Higher link availability is possible using lower data rates.

Data rate versus range is shown in figure 4.3.2-3 for the critical Mars-to-Earth links. Indicated are the point designs, which provide 10 Mbps and 20 Mbps (normal mode) capability at maximum range for the relay satellite and Mars piloted vehicle respectively. The assumed maximum range for the Expedition case study is 1.6 au for the relay satellite and 1.8 au for the piloted vehicle; maximum range for the Evolution case study is 2.5 au. Figure 4.3.2-3 also depicts spacecraft-to-Earth range for various flights of the Expedition and Evolution case studies. The data rate can be increased significantly for shorter ranges during these flights.

Figure 4.3.2-4 illustrates a typical example of the trade space considered in developing the Mars telecommunications point designs. Required downlink transmit RF power and pointing accuracy are presented as a function of data rate and antenna diameter for the relay satellite and Mars piloted vehicle (Evolution case study). Reasonable transmit power/antenna diameter trade space is identified. Hatched areas show transition of different design approaches. Design points are selected considering applicable antenna technologies, complexity, antenna size impacts on spacecraft packaging, associated antenna pointing accuracy, and transmitter power requirements and available technologies.

Navigation. The navigation design for Mars missions consists of three complementary segments: an Earth-based navigation system (Segment I), an on-board navigation system (Segment II), and a Mars-based network navigation system (Segment III).

Segment I is the current Earth-based navigation system for robotic planetary exploration, updated with all available state-of-the-art developments at the time of the mission. This system can adequately support all phases of the mission except cases where real-time navigation is required, such as aerocapture, landing, or emergencies. The Earth-based navigation system constitutes the baseline from which the mission navigation system evolves, and it provides capabilities for testing, calibrating, and backing up the other two navigation segments. Technology for this segment is available.

Segment II is an onboard navigation system that satis-

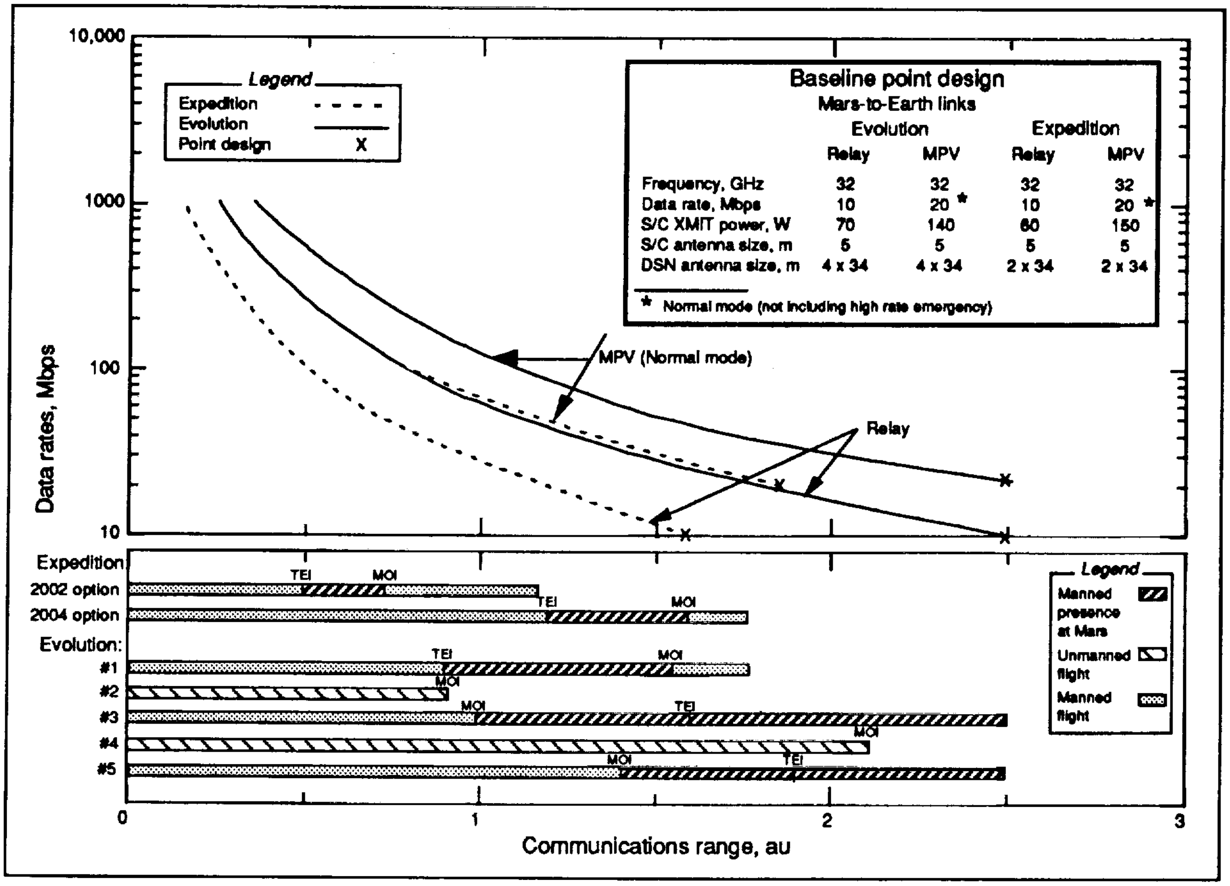


Figure 4.3.2-3.- Telecommunications data rate versus range — Mars-to-Earth links.

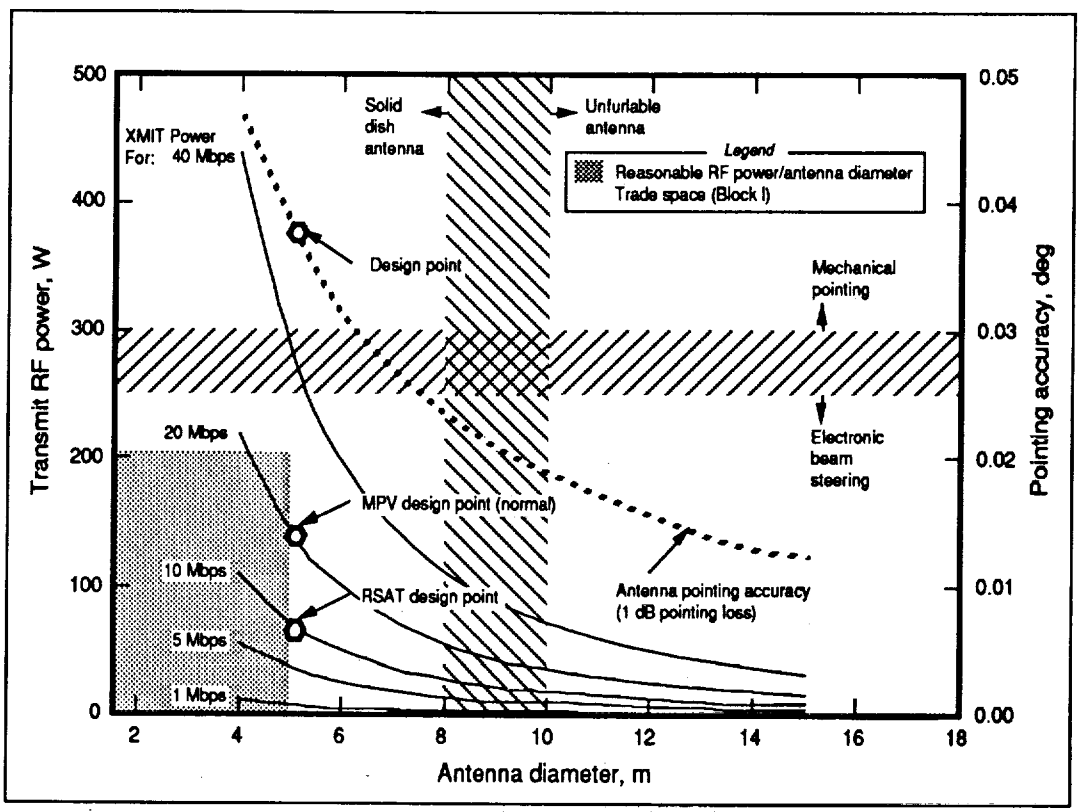


Figure 4.3.2-4.- Relay satellite, MPV transmitter power and antenna size trade; downlink to Earth (32 GHz) - Evolution case.

fies the need for real-time navigation of critical operations (e.g., aerocapture and landing). Navigation beacons will be deployed within 250 to 500 km of the landing site to assist in providing real-time navigation during landing operations.

Segment III, a Mars-based network navigation system, evolves in several phases. In the first phase, it consists of an adjunct of the two communication relay satellites to provide relative range and doppler data. During the second phase, global navigation beacons/transponders are deployed on Mars, Phobos, and Deimos. The habitat and the rovers are equipped with similar beacons or transponders for relative range or doppler data. A navigation center is established in the habitat with tracking and computational facilities to provide TDRSS-like navigation support during emergencies. In the third phase, several navigation satellites are deployed about Mars in two or more stages to provide real-time GPS (Global Positioning Satellite) type of navigation. This Mars-based navigation network has a role similar to that of the Segment I Earth-based navigation system, but for real-time operations in the martian system. Segment III not only provides assistance to the onboard navigation system for highly accurate real-time navigation, but it can also independently provide full navigational or computational support to any vehicle during emergencies, using the facilities available at the navigation center located in the habitat on Mars.

Table 4.3.2-I summarizes the assumed navigation performance requirements and estimated capabilities for the Mars missions. The delivery requirements for aerocapture are derived from Draper Laboratory studies for the Mars Rover/Sample Return mission, with the addition of 100% margin for piloted flights. The delivery requirements allow for possible fluctuations in the atmospheric density by a factor of two (either high or low).

Hazard avoidance and the need to land in close proximity to the habitat or other surface elements of the mission are assumed to require a landing accuracy of 10 m (3 standard deviations) or better. In order to achieve this accuracy, real-time tracking of beacons and/or a succession of optical landmarks, the location of which can be mapped to the landing site, is required. Since trajectory correction is not assumed to be available during the parachute phase, the estimated dispersions at the start of final powered descent require that the descent engines have a horizontal traverse capability of about 2 km during a period of 60 seconds.

Earth-based navigation, using radiometric data and optical position data for Deimos and Phobos, provides

TABLE 4.3.2-I.- MARS NAVIGATION PERFORMANCE ESTIMATES SUMMARY

Item	Navigation requirement	Navigation capability	Navigation segment/data types
Delivery error for aerocapture (3σ)			
Vacuum periapsis altitude, h_p:	±4.5 km	±3-5 km	Seg I (Earth-based navigation)/ radio and optical data [for initialization]
Flight path angle at entry, γ	±0.4°	±0.25°	Seg II (onboard navigation)/ optical data and ranging to relay satellites
Mars landing error (3σ)			
Position at landing:	10 m	5-6 m(1)	Seg II (onboard navigation)/ IMU and landmark tracking or beacons
Mars orbit error (3σ)			
Spacecraft position:	1,000 m (Routine ops)	300-700 m	Seg I (Earth-based navigation)/ radio and optical data
	100-300 m (Critical ops)	150-450 m	Seg III (onboard navigation)/ S/C ranging/doppler with beacons, etc.
		100 m	Seg III (block II enhancement/ GPS-type data)
Surface relative location error (RSS)			
Rover to habitat separation: (50 km nominal)	30 m	25 m(2)	Seg II (onboard navigation)/ doppler with 1 orbiter

(1) Assumes descent engine horizontal traverse capability of about 2 km in 60 sec

(2) Assumes favorable geometry

adequate accuracy for routine operations in Mars orbit. As indicated in table 4.3.2-I, Mars orbit navigation capability can be progressively enhanced by the phased deployment of a Mars-based navigation network system (Segment III).

Information Management. The Mars information management system consists of a distributed multiprocessor network with nodes located on selected flight elements as well as elements on Earth. A simplified architectural configuration of the proposed system is shown in figure 4.3.2-5.

The entire data system functions as one distributed system. All nodes store data in standard formats; all interfaces use established standards. Any user may query data from any node. Unattended expert systems manage and move data and messages within the system; automatic path selection, message switching, and data buffering are provided. It is modular to support evolutionary development and growth. Data are processed and stored appropriate to the type; voice on magnetic tape, video on tape or laser disks, and telemetry for science and local Mars centered data achieves the necessary accuracy and represents a reasonable balance between onboard- and Earth-based navigation systems.

The major components of the information management system are the library, the post office, and the utilities. The library provides generalized storage, analogous to a modern public library, including permanent archive storage, mid- and short-term buffering, and multi-user access. The post office provides the data movement functions, including message priority resolution at telecommunications gateways. The utilities provide the various multi-user functions for accessing the library and post office, and for transmitting data to multiple users when necessary. User-supplied processes that need to communicate with one another will be encouraged to do so directly, bypassing the system when practical to avoid excessively burdening the system. Messages that must be carried by the telecommunications system would, however, need to be processed through the information management system and would be subject to priorities in gaining access to the telecommunication links. Human interfaces are accommodated through the user functions, and not directly to the system.

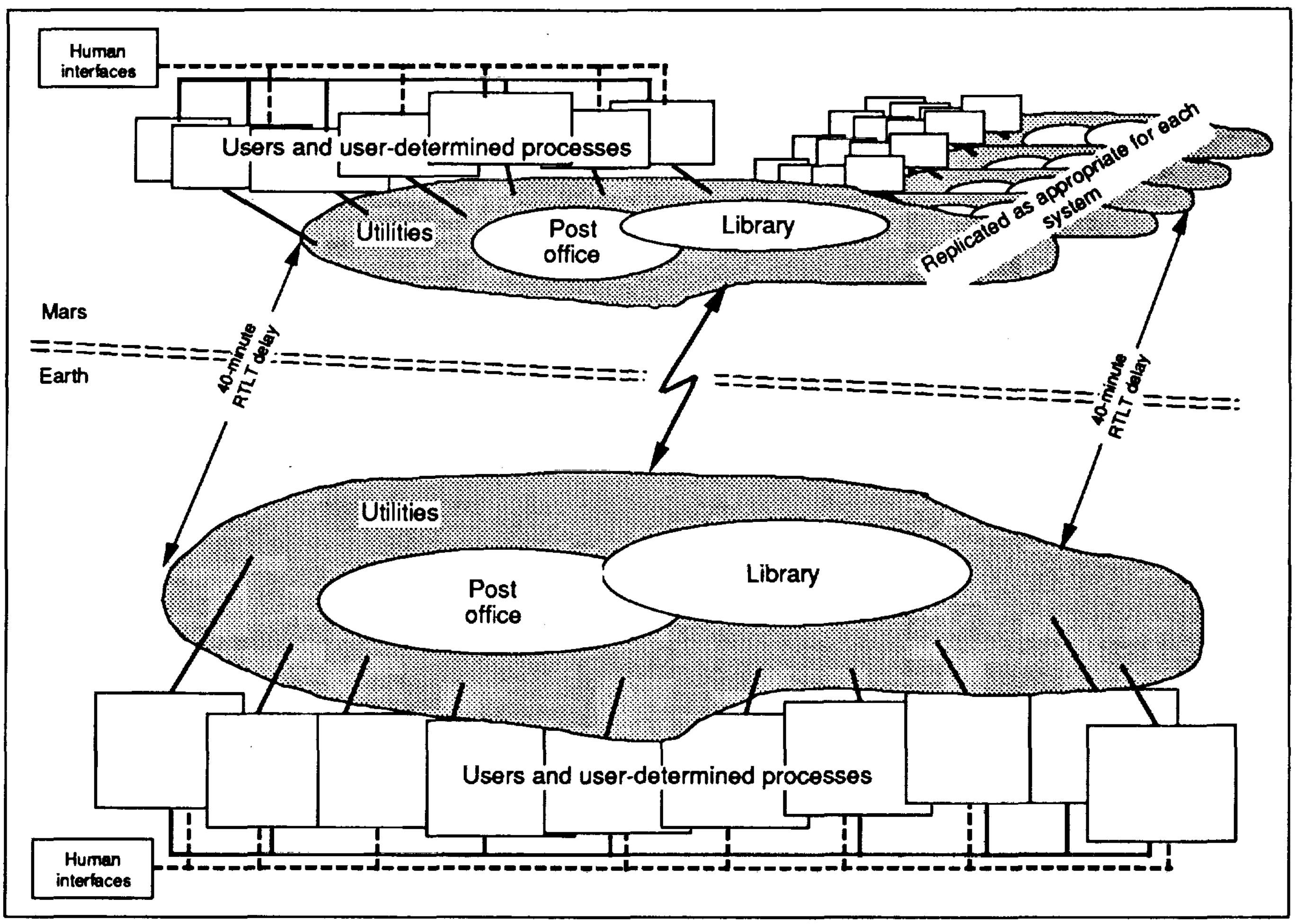


Figure 4.3.2-5.- Initial approach to Mars information management system.

Successful implementation of the information management system requires a significant degree of commonality in the interfaces with user-supplied processes. The system should provide the major design constraints on these interfaces, with user-supplied processes complying with the defined standards. Considerations of implementation cost, complexity, and maintainability make it desirable to implement as much of the system (and user-provided functions) on the ground as possible, consistent with a reasonable degree of spacecraft autonomy. However, response time requirements force many functions to be implemented in the spacecraft. Robustness, as opposed to reliability, requires that all key functions can be performed in diverse ways. Since data system functions are subject to systematic design errors, the mere replication of identical software in replicated hardware is not sufficient to ensure a robust system.

4.3.2.4 Mars TNIM Critical Technology Needs

The key technology areas required for Mars TNIM implementation are identified below. Descriptions and status of these technology needs are contained in section 4.3.3.

a. Ka-band technology to provide at least 10 Mbps at 2.5 au, as well as multi-beam, local Mars rates up to 50 Mbps

b. Data compression technology of at least 10:1 to provide real-time video, and concomitant link coding for $\leq 10^{-9}$ bit error rate

c. Spacecraft memory capacity technology for up to 10^{12} bytes to reduce channel peak loading

d. Onboard navigation and guidance system engineering demonstrations for aerocapture

e. Unattended operations

4.3.2.5 Development and Implementation Plan

The TNIM development and implementation sequence is summarized in figure 4.3.2-6. The critical schedule line items are (1) the architecture and system design development process, (2) interactions with Integration Agents to determine the designs and interfaces that are achievable from reasonable tradeoffs of technology, (3) the mission operations approach and requirements, (4) the process for identifying the needed flight/ground

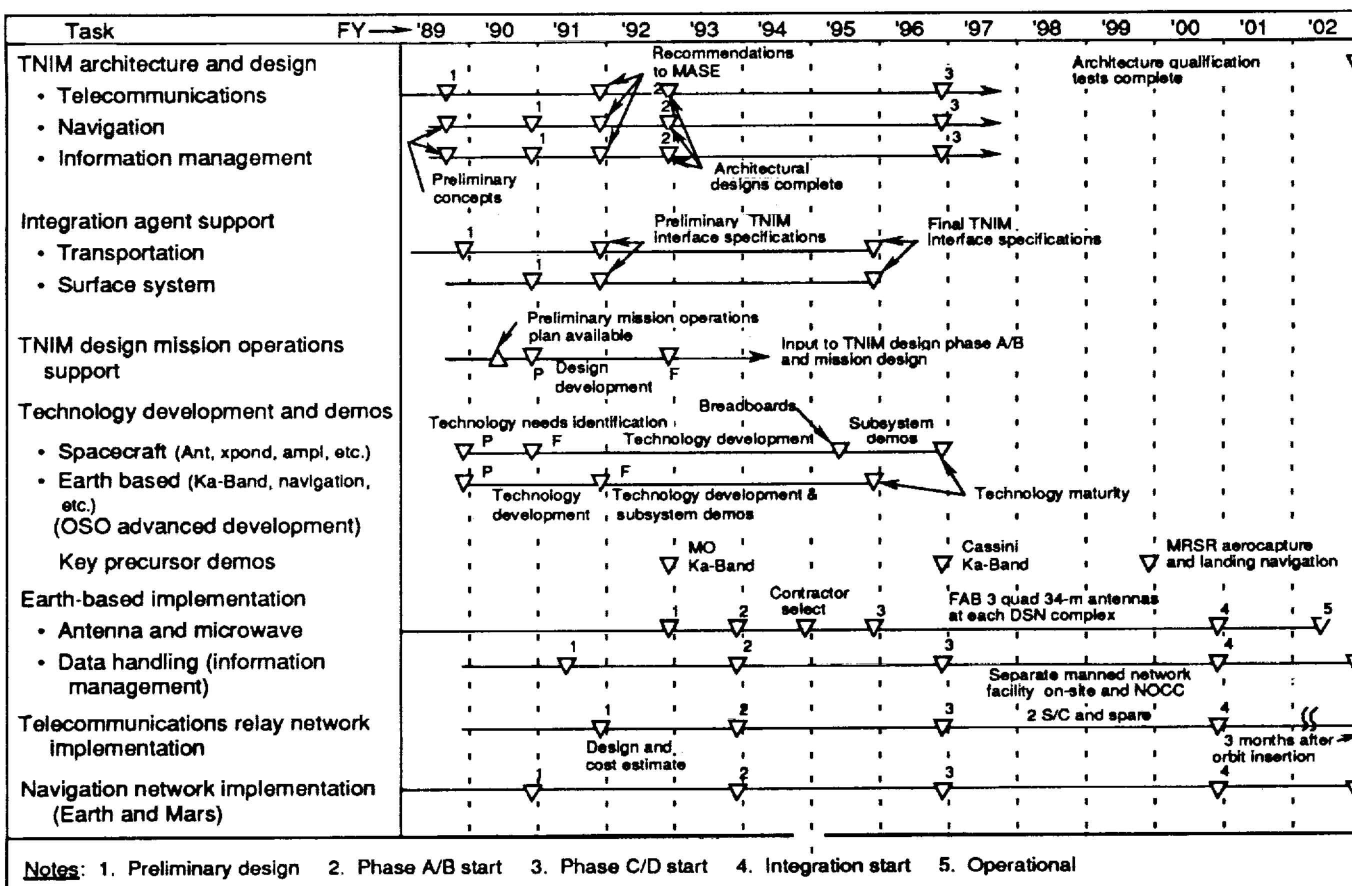


Figure 4.3.2-6.- Mars TNIM development and implementation schedule.

technology development, including selected advanced engineering model demonstration tasks; objectives for precursor missions, such as aerocapture navigation design validation, are set and scheduled to assist the TNIM development process, and (5) implementation of the Earth and space-based TNIM systems.

Seven significant milestones are: (1) completion of the TNIM architectural design by the end of 1992, (2) the national decision to pursue a specific objective, (3) preparation of functional and interface specifications for the Earth-based facilities, Mars relay network, and information management designs in 1995, (4) achievement of technology maturity through development and precursor demonstrations by 1996 to 1998 (later precursor demonstrations can be useful, but may require costly redesign and delays), (5) initiation of Phase C/D start in 1996 to 1998 (minimum implementation period is 5 years), (6) TNIM integration testing in 2001, and (7) relay network in-orbit certification 3 months after the first mission arrives at Mars in 2003 to 2005 (see precursor needs in section 4.3.2.6, below).

4.3.2.6 Design Conclusions and Issues

Mars mission operations needs are not yet defined to either specify real-time video (10 Mbps) from Mars, or to define real-time in situ missions operations functions at Mars for information management design; however, several existing information management architectural precepts may assist in focusing the requirements.

Ka-band technology is not presently available to support 10 Mbps Mars-to-Earth data rates. (Operational optical links will probably not be available until after 2010.) Mature Earth-based Ka-band technology is scheduled to be available by 1996. The corresponding maturity date needed for flight technologies for Ka-band, 10:1 data compression, and onboard storage (to reduce peak to average link loading) is 1996-1998. Those required flight technologies do not yet have sufficiently high development priority to be achieved.

Adaptation of 34-m DSN-type antennas in arrays for downlink reception and use of multiple uplinks appear to be adequate to meet Earth-based telecommunications and navigation mission needs; a dedicated manned-mission Earth subnetwork is proposed.

A Mars telecommunications relay network is required to provide high connectivity (>50%) links between Mars and Earth and between in situ Mars mission terminals.

The technology is probably available to support Mars aerocapture and landing navigation; establishing the atmospheric scale height and variability is probably the most critical technology need to assure an effective aerocapture design.

Precursor needs for Mars mission TNIM include (1) demonstration of a deep space Ka-band link, (2) use of minimum capability Mars telecommunications relay satellites prior to deployment of high performance satellites on the first mission, (3) advanced engineering demonstrations of aerocapture and landing navigation, (4) emplacement of radio beacons for lander support, (5) modeling of atmospheric scale height, and (6) measurements of surface conductivity for potential high-frequency surface-to-surface radio propagation designs.

4.3.3 Technology Assessment and Needs

Overall architectures and specific point designs to accomplish the TNIM functions for human lunar and planetary exploration have been determined. Lewis Research Center has examined these designs (described earlier) with respect to technology readiness and alternatives to identify technology needs. Telecommunications has received the most emphasis of the three areas under TNIM during 1989; technology alternatives and needs in this area have been determined and evaluated. Navigation and information management have received much less emphasis during the year. For these two areas, technology alternatives and needs have been examined at the functional level. This section of the report identifies technologies necessary to support TNIM for human exploration.

4.3.3.1 Technology Issues

Technology issues associated with telecommunications fall into three categories: (1) link data rate capability, (2) frequencies of communications links, and (3) transmitter/antenna system technologies. Tradeoffs can be made among user requirements, data compression, transmitter size, and antenna sizes; choices will impact the mass and power of the communications system.

For communications links, frequencies varying from UHF to Ka-band have been used in point designs. Frequency selection is most critical for the links back to Earth. For Mars communications, consideration of reasonable mass, size, and power to support the required data rates has led to the choice of Ka-band at frequencies near 30 GHz. Optical communications can have very high data rates, but pointing accuracy and stability problems are introduced. Additional study of radio frequency and optical tradeoffs is planned.

The third telecommunications category is transmitter/antenna system technologies. Major elements in establishing the required communications link capabilities are the transmitters and antennas. Technology advancements continually alter tentative choices of solid-state

power amplifier (SSPA) or traveling-wave tube amplifier (TWTA) transmitters. Technology advances make array antennas a realizable alternative to reflector antennas. Array antennas with multibeam or scanning beam capabilities are alternatives to steerable antennas. Array technology advances may contribute to lower mass and power transmitter systems. Array/reflector antenna tradeoffs are planned.

Technology issues associated with navigation are accuracy and autonomy. Order-of-magnitude improvements are required in navigational accuracies, and navigation autonomy is necessary where round-trip communications time exceeds limits for effective control. Aerocapture navigation requires both extreme accuracy and a high level of autonomy. The effect of the martian atmosphere on communications during entry is critical; this area must be studied and modelled to assess its impact on navigation communications.

Regarding information management, data compression offers the greatest opportunity to reduce both link sizes and data storage requirements. With video sources providing most of the high data rate needs, image data compression is of the highest priority.

Mass data storage prevents the loss of data from link interruptions (both planned and unplanned). The high capacity (terabytes) data storage needed requires storage technology development effort.

Elements of a "switchboard in space" can contribute to reducing life-cycle costs. With long round-trip delay times, onboard processing can provide efficient use of the communications system via autonomous routing and reconfiguration. Onboard processing may also include automated network control for scheduling and system management.

4.3.3.2 Lunar/Mars Technology Differences

The TNIM architectures for lunar and Mars case studies contain a great deal of similarities, but substantial differences translate directly into different technology needs. The Moon-to-Earth path length results in a transmission delay of approximately 1.25 seconds, but from Mars, the delay can be as long as 20 minutes. The other major difference is the geometry: the Moon orbits Earth, and their relationship to each other is relatively fixed; both Mars and Earth revolve around the Sun, and their relative positions change constantly.

The longer path from Mars to Earth requires greater transmitter power and larger antenna dimensions to provide communications. For a given transmitter power and antenna, a link from Mars to Earth cannot support as high a data rate as that from the Moon to Earth. In order to compensate for the lower data rate, TNIM designs for Mars will require extensive use of data compression techniques and data storage. By using data compression, information can be transmitted at a lower data rate, but with added complexity. Data storage can reduce real-time data rates by reducing the peak demand and forwarding data at a later time when demand is low. Unfortunately, this approach introduces a time delay. Alternatively, an optical system could provide the desired data rates for the Mars-to-Earth link, but this technology is less advanced than Ka-band. The greater distance causes the most significant differences in the area of navigation. In the lunar case, the navigation system could be Earth-based, whereas the Mars navigation system, also Earth-based, will require a space-based real-time autonomous structure for key activities such as aerocapture and landing.

4.3.3.3 Telecommunications Technology

An outcome of identifying TNIM system architectures and technology issues is the identification of technology needs, indicating where research and development should focus to enable these missions. In the telecommunications area, Ka-band communications technology has been identified to meet the mission requirements, whereas optical communications technology has been identified as an alternative if the data rate requirements increase significantly. Point designs for the 1989 Mars and lunar case studies have identified the use of Ka-band frequencies (18 to 40 GHz) to meet the majority of the telecommunications requirements. In order to meet the technology readiness level for the proposed time frame and to provide the level of reliability necessary for manned exploration missions, technology development has been identified in the areas of Ka-band transmitters, antennas, and monolithic microwave integrated circuit applications. A detailed discussion of these technologies follows, and a summary is given in table 4.3.3-I.

Ka-band transmitters, both traveling-wave tube and solid-state power amplifiers, are at relatively low technology readiness levels and will require further development to meet the exploration time frames.

Traveling-wave tube amplifiers can achieve higher power levels than their competitor, solid-state power amplifiers, and they are highly efficient; the TWTAs can easily support broadband transmissions, and high reliability has been proven in space applications. However, TWTAs require complex high-voltage power supplies, are not as efficient at low transmit power levels (6 to 10 watts), and in general, are relatively heavy.

Solid-state power amplifiers, on the other hand, operate at lower power levels of 1 to 20 watts and do not require high-voltage power supplies. These amplifiers also have

TABLE 4.3.3-I.- TNIM TECHNOLOGY STATUS CHART

TNIM technology	Technology requirement	Technology readiness level
Ka-band TWTA transmitters	10-150 W high-efficiency	3-4
Ka-band SSPA transmitters	1-15 W high-efficiency	2-3
Multibeam antennas	10-20 simultaneous beams	5
MMIC multibeam antennas	10-20 simultaneous beams	2
Direct-radiating MMIC phased array	10 x 10 array	3
Reconfigurable antennas	Operation at Ka-band	2
MMIC technology	Higher power-lower noise	3-4
Optical technology	Higher power laser source	3
Data compression	10:1 compression without loss	2
Data storage	10^{12} byte storage capacity	2

the potential to provide a higher reliability, although this has not yet been demonstrated. Finally, SSPAs have the potential to be lower in cost, and they are amenable to phased-array applications.

To meet the antenna requirements of specialized coverage areas and tracking needs of TNIM missions, specialized beam-forming architectures may need to be developed. Areas to be investigated for each proposed link are scanning multibeam, steerable, direct-radiating phased array, and reconfigurable antenna technologies, as outlined in table 4.3.3-I. Requirements for lunar and Mars communications links are rather stringent.

Monolithic microwave integrated circuits (MMIC) are circuits in which all active elements and their associated passive elements and interconnections are formed into the bulk, or onto the surface, of a semi-insulating substrate by semiconductor processing techniques such as epitaxy, ion implantation, sputtering, and evaporation. Many antenna applications, such as multibeam and phased arrays, can benefit through the development of this technology. MMICs offer the potential for improved performance, higher reliability, radiation hardening, and size/weight reductions.

Although the current point designs employ Ka-band frequencies, the potential advantage of optical communications over RF systems cannot be overlooked. Optical communications have a potential gain of 60 decibels over Ka-band, which can translate into reduction of the transmit/receive apertures, savings in spacecraft power, and improvement in link performance. Optical communication could support data rates on the order of 300 Mbps for Mars-Earth distances. However, optical communications technology presents greater development challenges than Ka-band. Development is necessary in the areas of long-life, higher-power laser sources, photo detectors and detector arrays, accurate telescope pointing, and acquisition and tracking technology.

4.3.3.4 Navigation Technology

The lunar and Mars TNIM systems have different technology needs. Navigation for lunar case studies is fairly simple in comparison to the Mars case studies. The lunar navigation system utilizes the Earth ground terminal to perform range and range-rate extraction of the transit user signals relayed by the geosynchronous relay satellite. The ground terminal also performs orbit/position determination for lunar users and Office of Space Operations system elements. Precision orbit determination of scientific lunar orbiters for mapping or gravitational field measurements requires L2 relay satellites (farside). Navigation for the Mars case studies is performed in three areas: (1) an Earth-based system that is responsible for initial routine navigation support, (2) the onboard system that provides a basic level of "near" real-time onboard navigation capability that supports approach and aerocapture, landing, surface exploration, ascent, and rendezvous and docking, and (3) the Mars-based navigation system that provides orbital navigation and real-time navigation support for all operations in the martian system.

Key navigation requirements for the missions will drive

technology development efforts. Although delivery error for aerocapture is the most critical requirement, achieving Mars landing accuracy of 10 m or better and spacecraft position determination within 1 km for routine operations and 100 to 300 m for critical operations will also require technology development.

4.3.3.5 Information Management Technology

Current studies indicate a definite need for data compression, especially for the Mars missions, to reduce real-time high data rate requirements. Data compression can also reduce power and/or antenna requirements for high data rate links. There are two primary areas for application of data compression algorithms: video transmissions and science data. In both areas, lossless data compression of at least 10:1 is a requirement for the lunar and Mars TNIM architectures. Once candidate data compression schemes are chosen for implementation, software and/or hardware must be developed to minimize the complexity, weight, and power impact of implementation on the systems.

Data storage reduces real-time transmission rates and is required in both the lunar and Mars exploration missions. Storage can be used as an online buffer for intermittent, high-rate transmissions and to prevent loss of low-rate data during emergency communications. Additionally, mass storage is needed to store data during periodic outages or unavailability of data links. Current Mars studies indicate development is needed in mass data storage media and devices to achieve a storage capacity on the order of a terabyte (10^{12} bytes) or more. Additionally, input/write rates on the order of 300 to 500 Mbps are required. Technology development is needed in the areas of magnetic and optical storage media and devices since current technologies fall far short of these requirements.

4.3.3.6 TNIM Technology Development Plan

The plan shown in figure 4.3.3-1 depicts a timetable for research and development of the identified technologies designed to meet the proposed Lunar Evolution and Mars Evolution mission time frames. The milestones represent an initial study period in which alternative designs are evaluated to result in a final design. This is followed by the development of a proof-of-concept (POC) hardware implementation. Space qualification and, where necessary, flight demonstration, brings the development to a technology readiness level 7. The plan assumes reasonable, appropriate funding and results, where possible, in a technology demonstration in the year 1997 corresponding to the Mars Evolution Phase C/D start-up.

4.3.3.7 Conclusions

Examination of the technology readiness and alternatives for the TNIM areas has identified Ka-band communications as the critical technology for telecommunications, with transmitters and antennas the most critical elements. Continued development of optical communications technology could provide an alternative, since

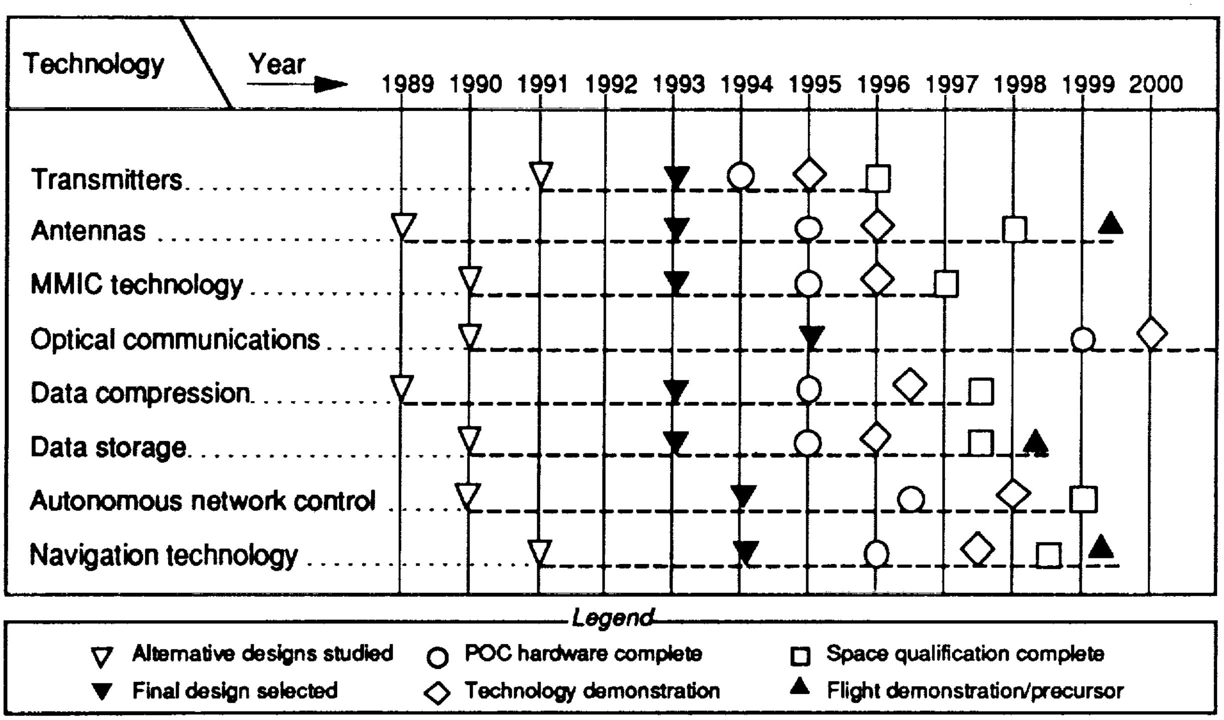


Figure 4.3.3-1.- TNIM technology development plan.

data rate requirements may increase. Relay satellite systems are needed for both telecommunications and navigation. For navigation, the most critical technology need is in aerocapture navigation and guidance. Additional elements requiring development are navigation satellites, sensor technology, and onboard computers for autonomous operations. For information management, the critical technologies are data compression and data storage. Image data compression is needed to reduce the data rates and to lessen the data storage requirements. Large capacity mass data storage is needed to meet mission requirements and needs development. Mission timetables demand funding and initiation of effort in FY 1990 in the identified technology areas.

SECTION 5

Preparatory Program Descriptions

Part of the development and analysis of a range of alternatives for expanding human presence beyond low-Earth orbit is the integration of the other NASA programs' functional capabilities and responsibilities that are prerequisite to the implementation of one or more exploration alternatives. Therefore, OEXP has provided requirements for human exploration to the various affected NASA program offices in the form of the Study Requirements Document (SRD). This section describes the current NASA preparatory programs in life sciences, robotic missions, and technology development.

5.1 LIFE SCIENCES

Human exploration of the Moon and Mars presents unique challenges for life sciences and life support beyond those required for the Shuttle or Space Station Freedom, and life sciences programs significantly impact the ability of the United States to undertake the exploration missions. The physiological and psychological effects of extended exposure of humans to space radiation, artificial gravity, and isolated and confined environments must be understood.

Since its establishment, the Office of Space Science and Applications Life Sciences Division has had two primary goals: to assure the health, safety, and productivity of humans in space; and to acquire fundamental scientific knowledge concerning space life sciences. These two goals are interactive and have been carefully balanced in the life sciences research programs. The first goal is mandatory for supporting the Agency's exploration goals. The second goal supports the first and is a critical component of NASA's balanced space science program.

Potential contributions by international partners and the U.S.S.R. are not factored into this document. Nonetheless, such contributions can be substantial, and the Life Sciences Division plans to pursue the international cooperation that is already in place for Spacelab, and planned for Space Station Freedom.

5.1.1 Approach to Meeting Human Exploration Requirements

To conduct the exploration missions described in the OEXP case studies, the following five life sciences areas are required:

1. **Advanced Medical Care:** Provide for remote medical care in the event of illness or injury, develop methodologies for crew health maintenance and monitoring, and identify crew medical skills requirements.
2. **Reduced Gravity Countermeasures:** Develop methods to maintain the health and physical capabilities of crews during exposure to micro-, reduced, or artificial gravity and to facilitate readaptation to Earth's gravity.
3. **Radiation Protection:** Determine chronic low-dose, solar flare, galactic cosmic rays, and other space and manmade radiation risks and develop appropriate countermeasures and warning capabilities.
4. **Life Support:** Develop processes for space transfer vehicles, planetary surface outposts, EVA space suits, and planetary roving vehicles to revitalize air and water, supply food, and monitor and decontaminate the environment; in addition, develop the protection and warning requirements associated with other hazards, such as toxic gases and micrometeoroids.
5. **Space Human Factors:** Optimize systems design requirements, procedures, and measures to ensure safe, productive, and enhanced crew performance.

The importance of each of these areas varies according to the particular mission. The amount of information that remains to be gathered before an area can be acceptably addressed for each mission's conditions also varies.

5.1.1.1 Advanced Medical Care

Advanced medical care addresses specific capabilities that are critical for monitoring and maintaining crew health for extended-duration missions and for sustaining a high level of performance and productivity both in transit and on a planetary surface. Advanced medical care for exploration missions will require enhanced in-flight medical capabilities, on-board medical expertise, and current clinical techniques. Performance requirements for all in-flight medical hardware and software will need to be developed, and trade studies (e.g., weight, power, consumables, in-flight maintenance, and cost) will need to be conducted to determine optimal systems.

Portions of a Health Maintenance Facility (HMF) are under development and will be flown and evaluated on early flights of Spacelab and the Extended Duration Orbiter to eventually become operational on Space Station Freedom. The HMF will provide onboard diagnostics, therapeutics, monitoring, and medical information management. An advanced medical care facility is also planned as part of the Advanced Technology Development Program for Space Station Freedom follow-on.

These facilities are expected to meet near-term needs for the health care of astronauts.

As applied to exploration missions, these facilities should prove adequate for the Lunar Evolution case study, and with significant enhancements, for the Mars Evolution case study. In the Lunar Evolution case study, the relative proximity of the astronauts to Space Station Freedom and the possibility of quick, although not immediate, rescue and return to Earth lessen the need for more extensive facilities. Nevertheless, on the lunar surface there will be a greater need for surgical and intensive care unit capabilities than on Space Station Freedom. This demand means that on-board computer-aided diagnosis systems, automated, miniaturized clinical chemistry systems, and a general surgery capability will be necessary. An evaluation will be needed to determine the scope and design of an autonomous HMF and also to ascertain the skills required to make the operations of such a facility practical.

Space Station Freedom technology will provide a basis for a lunar operation. A lunar outpost could provide the reduced gravity environment in which to assess the requirements and develop the approaches necessary to meet the greater needs anticipated for a Mars mission. Without the ability to return crews quickly to Earth, a Mars mission facility will be required to provide for a wider spectrum of contingencies. In all mission scenarios, the capability to provide medical care will be defined by medical support equipment and by the medical skill of the crew; this capability is a key factor for the exploration missions.

5.1.1.2 Reduced Gravity Countermeasures

Exposure to reduced gravity leads to a significant number of physiological changes in humans, including:

a. Negative calcium balance resulting in the loss of bone

b. Atrophy of antigravity muscles

c. Fluid shift and decreased plasma volume

d. Cardiovascular deconditioning resulting in orthostatic intolerance

e. Reduced tolerance to increased gravity on reentry

f. Possible changes in the body's immune system

In addition to these direct physiological effects, microgravity also leads to changes in neurosensory function, which, in about 40 to 50 percent of individuals exposed for the first time, leads to space motion sickness, part of the so-called space adaptation syndrome. Although it appears that humans can adapt to the neurosensory changes, the physiological effects are progressive and, with extended periods in space, tend to be cumulative and require intervention techniques or countermeasures. However, just what combination of countermeasures and procedures is necessary or preferred under what flight conditions is yet to be determined.

So far, the U.S. and U.S.S.R. have relied on exercise regimes, usually vigorous and protracted, to provide the desired protection. However, it is not clear if physical exercise will be capable of maintaining crew health for very-long-duration missions. Astronauts may find difficulty keeping up the required exercise programs for the duration envisioned for most exploration missions. If an astronaut should suffer an accident or become ill and be unable to exercise for a protracted period, severe deconditioning would result.

The Life Sciences Division plans to use the opportunities provided by Space Station Freedom to enhance current knowledge on the biomedical effects of weightlessness, particularly the extended-duration effects. The planned Biomedical Monitoring and Countermeasures program on Space Station Freedom will provide countermeasures for 6-month crew exposures to microgravity. This duration extends by more than 3 months the longest previous American flight, Skylab 4. In addition to enhancing biomedical knowledge on the effects of prolonged weightlessness and the efficacy of exercise in maintaining conditioning, the program on Space Station Freedom will allow for the testing of alternative or supplementary countermeasures such as diet, pharmaceuticals, and perhaps electrical stimulation of muscles. In order to determine whether zero gravity is an acceptable environment for the MTV (300- to 400-day nominal mission), progressively longer tests will be conducted on orbit. The plan is to provide the requirements for the MTV by Phase B of the vehicle development.

Space Station Freedom will also allow initial assessments to be made with animal subjects of the effects of artificial gravity. Although artificial gravity has never been tested in space with humans, it has been proposed as a technique for managing the long-term effects of weightlessness. Much as the presence of gravity alone does not guarantee fitness on Earth, artificial gravity alone is unlikely to prevent all bone loss, muscle atrophy, and changes in the cardiovascular system associated with the space environment, and it may introduce vestibular symptoms. Nonetheless, a carefully developed regimen of artificial gravity combined with exercise could provide the desired results.

The need to provide artificial gravity either continuously or intermittently during exploration missions is based on a series of assumptions that need to be validated. The first and most critical assumption is that all adaptive

changes of the body to microgravity can be provided by physical activity in a constant exposure to acceleration equivalent to that of the Earth's gravitational field (1 g). In addition, it is assumed that physical activity plus fractional g, such as that encountered on Mars (0.38 g) or the Moon (0.17 g), may prevent or reduce physiological deconditioning. It is also assumed that as gravity load increases, the requirement for physical activity decreases.

Technology to accurately measure the actual g loads imposed on the body during the course of normal daily activities (particularly on the lower extremities) must be improved so that the actual physiological g-loading needed to maintain normal functioning can be specified. Simulation models of the effects of weightlessness have proved to be of value in approximating the physiological deconditioning of space flight. Bedrest simulation studies will be used to determine the minimum time of exposure to 1 G_z (without or with activity) required to prevent changes in the cardiovascular, musculoskeletal, and nervous systems.

Artificial gravity can be produced in weightlessness either by rotating the entire spacecraft or by carrying a human-rated centrifuge along with or on-board the space vehicle. Unique impacts are associated with providing artificial gravity, and its use may be indicated only if other solutions are found to be inadequate. A long lead time is required to determine if artificial gravity is effective or even necessary. Supporting requirements are provided through the Human Performance element of Pathfinder.

Construction and flight testing on Freedom of a research centrifuge facility is part of the artificial gravity research plan, and definition studies of human-rated artificial gravity devices should begin as soon as possible. The gravity threshold may be different for different physiological systems and different species. Animal flight experiments using the centrifuge facility to compare constant g exposure at different levels to that of intermittent g could provide information on the relative merits of constant versus intermittent gravity exposures. The direct relevance of these studies to man may be limited, and results should be interpreted with caution.

Equally important is the determination of the acceptable g-limits and the angular velocities that an onboard device or a particular vehicle design would impose. The adaptability of individuals to changing rotational environments must be determined. It will be necessary to develop a human-rated variable gravity research facility in order to obtain a complete understanding of artificial gravity and associated exercise countermeasures.

5.1.1.3 Radiation Protection

A significant life sciences support area to be addressed in the 1990s is crew protection from space radiation. The space radiation environment is complex and has serious implications in terms of acute and chronic effects. During space flight, crews are exposed to ionizing radiation from a variety of sources, including trapped radiation belts (protons and electrons), galactic cosmic radiation, and the potential for solar particle events. In low-inclination, low Earth orbit (LEO), the primary source of chronic radiation is the inner trapped proton belt. Some orbital inclinations will traverse the South Atlantic Anomaly, where the inner radiation belt is closest to Earth. In addition, in the polar orbits (at the north and south poles), there is less shielding by the geomagnetic fields and thus increased exposure to galactic cosmic rays and solar particles.

Missions in high-altitude geosynchronous Earth orbit (GEO) are subject to radiation exposure from electrons in the outer radiation belt, bremsstrahlung from electron shielding interactions, galactic cosmic rays, and solar particle events. Predicting dose rates in GEO is complicated by the wide variations in temporal intensities due to diurnal cycles and solar activity. Galactic cosmic radiation consists primarily of protons (87 percent), helium ions (12 percent), and high energy ionizing particles (1 percent). Based on data from previous research, exposure to ionizing radiation may be a limiting factor for both mission and career durations for space crews. In addition, periodically and unpredictably, solar flares can irradiate limited regions of space with high doses that include protons and heavier particles.

NASA is required to provide a system of radiation risk limitation for all astronauts and space workers using the ALARA (As Low As Reasonably Achievable) principle for radiation dose rates. The ongoing life sciences program includes space radiation protection research, yet additional resources are needed to fully investigate and provide space radiation protection. From a practical position, all radiation exposure risks could possibly be reduced, but not completely eliminated. The fundamental need is to reduce the uncertainties (2X-10X) regarding the galactic cosmic radiation environment. This radiation environment includes the area outside the spacecraft and the resulting dose inside the spacecraft and inside crewmembers' bodies, as well as the relative biological effectiveness (carcinogenic and mutagenic) of the various high-energy particles involved. Cancer and mutation risk estimates using conventional assessment procedures are inaccurate, and the projected dose-equivalent for a Mars mission exceeds the current astronaut radiation exposure limits. A more realistic assess-

ment procedure for galactic cosmic radiation needs to be developed from ground- and space-based radiobiological research results.

The approach to this issue is to utilize existing and new information on the biological effects of conventional ground-based radiations to assess space hazards, to generate new data on biological effects of high energy heavy ions and secondary particles, to develop additional dosimetry technology to provide accurate and reliable passive and active dosimetry for the actual radiation exposures experienced in space flight and for radiobiological purposes, to further develop models of the space radiation environment for lunar and Mars missions, and to develop and validate transport codes that may be used for accurately determining dose, dose-equivalence, and fluence through various shields of different materials. These transport codes must be flexible to support the evolving radiation risk criteria associated with exploration missions.

In terms of the external and internal radiation environments, it is critical to obtain accurate and reliable descriptions of the fluxes and types of primary and secondary particles. The Life Sciences Division is currently planning a reusable, free-flying biological satellite program (LifeSat), which will provide the capability to study the biological effects of radiation dosages, with Phase C/D currently scheduled for 1992. LifeSat will have unique technical capabilities to provide artificial gravity levels (0 to 1.5 g), long-duration missions (up to 60 days), and access to orbits (e.g., polar) that can target experiments to different radiation fields. LifeSat will provide accurate and reliable information on a unique spectrum of radiation that will be invaluable for risk assessment planning for lunar outposts and Mars missions. For example, it has been estimated that a 60-day LifeSat mission in polar orbit (90-degree inclination) would simulate 5 percent of a Mars mission. LifeSat missions will provide radiation information that cannot be obtained through ground-based experiments.

The most potentially catastrophic radiation problem for space crews, in terms of severe damage or even death, is the occurrence of a solar particle event. Nominal spacecraft thicknesses provide little shielding against solar particle event protons. Well-shielded radiation shelters must be provided on exploration spacecraft. Warning systems need to be developed that provide adequate notice of a forthcoming solar event. Currently, no theory exists for realistically selecting and assessing shielding materials and thicknesses. Current transport codes are used for computing the radiation environment. However, the uncertainties that exist in the computational environment result in intolerable uncertainties in the shield thickness required. Theoretical and experimental research is needed to reduce uncertainties and answer shielding questions such as material, size, mass, design, and structural integrity. Accurate flexible transport codes must be developed to support evolving risk criteria for exploration missions.

An important concern is the possibility that the condition of weightlessness, or reduced gravity, exacerbates the negative effects of radiation. During weightlessness, the effectiveness of the immune system in mitigating the impact of radiation exposure is reduced. If an interaction exists between microgravity and radiation, levels of acceptable dosimetry may need to be adjusted downward, increased protection may need to be provided, or some other means may need to be developed to lower the risk of space radiation exposure.

A long-term research program will examine the nature of the ionizing radiation environment in space and determine its implications for human exploration missions to the Moon and Mars. To assure the safety of human space exploration, specific program objectives include: develop accurate and reliable dosimetry measurements of cosmic and solar radiation; quantify various long-term biological effects of space radiation as a function of dose-equivalence, fluence, or a biological dosimeter end point such as chromosomal aberrations; develop accurate dosimetry for conducting radiobiological studies, monitoring astronaut exposures, and developing and testing models of the space radiation environment; develop countermeasures, including radioprotectants or nutritional supplements; provide adequate shielding; and develop a solar monitoring, detection, and warning system. A baseline ground-based research program will be augmented by flight experiments, especially in polar orbits, to study the effects of high-energy ionizing particles.

Specific shielding technologies to be evaluated and developed are stored water or propellants, new lightweight composite materials, active electromagnetic radiation shielding, and use of planetary surface materials for outpost habitats. As a means of reducing the biological impacts of radiation, selected pharmaceutical radioprotectants and dietary supplements also need to be evaluated.

5.1.1.4 Life Support

The development of regenerable life support systems with enhanced efficiency for food production and water and air recycling is required for long-duration missions with sizable crews but is beyond the current state of conventional life support technology. As mission duration and number of crew increase, the mass and volume required for consumable life support supplies and spare parts also significantly increase. The reliability of current systems will be taxed as the need for regen-

eration increases and logistics lines increase in length and complexity. A solution to the problem is to develop reliable recycling technology for life support needs.

There are three major applications for life support systems in human exploration missions: (1) space transportation vehicles, (2) extravehicular activity suits and vehicles, and (3) extraterrestrial habitats. The life support requirements for each include supplying life support needs and monitoring air, water, and food quality. The use of in situ resources to reduce life support logistics requirements will enhance long-term human habitation away from Earth.

For vehicle and surface EVA systems, an improvement in physical-chemical life support technology is needed across the board. There have been significant advances since the Apollo era, but existing portable life support systems are too large and require extensive logistical support. To meet the needs of the exploration missions and minimize the risks for surface EVA systems, lightweight, high pressure suit options need to be readied for advanced development. Areas to be emphasized will include: lightweight and durable materials, glove design, dust contamination protective measures and techniques, lower torso mobility systems for walking, ancillary mobility systems for surface transportation, and lightweight, compact, portable life support system technologies.

For habitats, a major life support objective for exploration-class missions is to develop a regenerative system through the integration of biological and physical-chemical processes. This system could reduce or even eliminate resupply problems by producing food, potable water, and a breathable atmosphere from metabolic and other wastes in a stable, efficient, and reliable manner. An important related near-term objective is to develop an accurate monitoring system to assess the quality of recycled air, water, and eventually food and to detect trends or changes in environmental quality.

Once improved regenerative systems have been ground-tested, Space Station Freedom will be used to assess reliability in microgravity, both at a subsystem and at an integrated-system level. Physical-chemical systems developed in the past are being integrated and tested by the Space Station Freedom program, but the final configuration of the Freedom Station life support system may not be fully regenerative, even for air and water. Advanced technologies must be made available early in the development of the exploration missions. Of particular urgency is the requirement to reprocess and reclaim wastewater. Water is essential, but massive and expensive to transport. A high percentage of water reclamation is a first-order objective being pursued through ground-based research. The technology needed to increase the regenerative yield is within reach, but at present it is uncertain whether the final Space Station Freedom configuration will incorporate water reclamation.

For the exploration missions, a series of research activities to develop a complete or partial bioregenerative life support system is underway and planned for later testing on Space Station Freedom. The first is a technology demonstration project for a regenerative life support facility that evaluates performance of plant growth technologies in the space environment. Once the facility is available, it will be used to determine the effects of the space environment on plant productivity across a range of environmental parameters. A final step in the demonstration series of regenerative life support projects will be to build, test, and evaluate the physical/chemical/biological systems required for future exploration missions.

Essential in the development of regenerative systems are investigations and technology development in the areas of plant growth, food processing, sensing, controls, and automated monitoring and contamination detection systems. Information from space human factors will be integrated with life support requirements for food preparation systems.

For any life support system, comprehensive environmental monitoring is required to help ensure crew health and safety. Possible chronic exposure of crewmembers to environmental contaminants can result from materials offgassing, undetected leakage of payload and utility chemicals, or malfunction of life support systems. Although general monitoring requirements, such as the need to monitor organics in the spacecraft atmosphere, may be constant across all exploration missions, some monitoring targets will be mission-specific. Specific contaminants may be different in space, on the Moon, or on Mars.

Environmental health system equipment being developed to monitor the quality of Space Station Freedom's air and water may be adequate to meet many environmental monitoring needs during exploration missions. However, mission-specific design constraints, such as limitations on consumables, will likely be more demanding for exploration missions than for Space Station Freedom. In this case, development of new monitoring technologies may be required to meet the environmental health needs of exploration missions.

5.1.1.5 Space Human Factors

For exploration goals, the space human factors research must expand from its current base to accommodate the new challenges associated with each mission. At present, it is the intent of the Life Sciences Division Space

Human Factors Program to develop a quantitative base for enabling human exploration missions, but further definition may alter the program substantially.

NASA's planned exploration missions will involve sending small groups of people into space for extended periods. The success of long-duration exploration missions involving either a lunar outpost, a Mars expedition, or evolutionary missions, will require a thorough understanding of the conditions that support and enhance human capabilities for living and working productively for prolonged periods of isolation and confinement. The requirements for designing environments and countermeasures that will ensure crew safety, productivity, and health are derived from the Space Human Factors Program in the Life Sciences Division. This program takes an interdisciplinary approach to understanding basic human capabilities in the space environment, including psychological, social, perceptual, and behavioral aspects, as well as such interactions with the physical environment as human-machine interface and habitability.

Future exploration missions will be carefully examined from a space human factors perspective. One of the key issues is the effect that prolonged isolation and confinement in artificial environments have on both individual functioning (i.e., psychological and behavioral) and crew effectiveness and performance. Significant questions concern crew composition and motivation, skill training and retention, inflight behavioral performance assessment, and communication and information management.

The social and psychological effects — the perceived quality of living — of the physical environment fall under the topic of "habitability." Spacecraft architecture and outfitting are particularly relevant. Studies are being planned to identify key habitability requirements, such as the amount of volume needed for individual and group activities, variety in the visual/aesthetic environment, work/rest schedules, personal health maintenance, hygiene, recreational facilities, and food preparation systems. These requirements will be derived through computer models in which movement and access are studied. These models will be verified in analogs such as underwater habitats and Antarctic research stations. Studies will also be performed in mockups and on Space Station Freedom.

Crew composition. To determine the composition of space-based and ground-based crews for long-duration missions, a number of factors must be considered: which personal and interpersonal characteristics relate to smooth-functioning and productive groups (baseline data), how the desirability of certain characteristics might vary with organizational variables such as task and authority structure, and what the impact is of sudden changes to group functioning, such as the rotation of members. Psychological issues related to the dynamics of crew composition are also important. These issues include interactions between crewmembers who explore the planetary surfaces and those who remain in the spacecraft, and psychological and social support for crews. In addition, research to examine crew psychological selection criteria will be undertaken for extended-duration, self-sufficient missions. Both simulation and field studies are being conducted, using the NOAA habitat, to provide significant information on baseline human factors questions. These studies are also examining the influence of task and authority structure as well as the influence of new members and unfamiliar crews on performance and productivity. Based on this research, crew training strategies have been developed and are being successfully implemented. As this work continues to grow and progress, it will be necessary to verify findings in a long-duration analog, such as Antarctica.

Training and Skill Retention. Training and skill retention will be studied in ground-based laboratories. One of the major issues for long-duration missions is cross-training, which is necessary because of the slow or nonexistent capability for quick return to Earth in the event of incapacitation of one crewmember and the limited crew size compared to the range of tasks that must be performed. A second key issue is skill retention. For example, in the Mars exploration mission, the crew must dock with Space Station Freedom 18 months or more after its last practice session. Onboard simulators and training exercises in the work schedule will be needed. These requirements will drive the design of workstations and onboard software, and must be identified early in the life cycle of the proposed programs. However, by definition, studies to measure skill retention over 18 months require a full 18 months to complete; thus, long lead-time testing is required.

In-Flight Behavior/Performance Assessment. Space environments, like other isolated and confined environments, are known to produce environmentally induced stress. In space, there are additional physiologically induced stresses. Environmental and physiological stresses in combination are likely to result in behavior/performance deterioration in long-duration space missions.

In-flight assessment of performance becomes as important for monitoring the crew's capabilities as built-in test equipment is for monitoring the craft's hardware status. Non-intrusive measures that are built into the software and hardware will allow evaluation of crew stress levels, error rates, frequency of utilization of various capabilities, and other measures of performance level. This information can be used to plan training and refresher courses, as well as to detect the need for other assistance

to the crews. Non-intrusive data collection procedures are currently being studied under the OSSA base program. These techniques need to be qualified in STS missions and on Space Station Freedom for use in planned exploration missions. In addition to the cognitive performance measures, physiological performance measures must be made. Models describing the effects of zero gravity and space suits on physical behavioral performance are being developed through the base program and the Pathfinder initiative. These models will provide a basis for systematic assessment of strength and motion capabilities in reduced gravity environments. In addition, performance assessment and exposure must be lengthened through long-duration analog studies. The results of these studies should be mapped to the results of the zero gravity/artificial gravity work to develop recommendations on total stress and its management in the space environment. In-flight experiments will be needed to validate findings and recommendations.

Crew-Machine Interaction. An additional performance factor is the interaction between crews and automated spacecraft systems. At present, well-established principles do not exist to guide the distribution of tasks between human and automated systems for maximum efficiency and reliability. Uniform assignments of tasks to crews and automated systems may increase problems caused by human errors or the inflexibility of automated systems. Performance measures can be examined through ground-based testing of new technologies, including reliability, feasibility, and utility. The psychological effects of automating critical systems need to be examined in high-fidelity environments where it is feasible to accurately assess efficiency, utility, and safety impacts. The Space Shuttle and Space Station Freedom can be used to measure crew use of and reactions to autonomous systems. Objective and subjective performance measures can be obtained by monitoring crew requests for information that the automated system does not supply, or obtained subjectively through crew feedback.

Communication/Information Resource Management. OEXP, through the NASA Headquarters Office of Space Operations, is currently implementing plans to meet the total communications and information delivery requirements implicit in the exploration mission. Over and above this basic requirement is the need to determine the dynamics of communication/information systems when functioning crewmembers are in the loop. Basic questions to be investigated include: What are tolerable delays in a two-way verbal exchange, and how long is the crew able or willing to wait for information? What are the implications of remote versus onboard information systems? How can information, at the right level and in the right form, be brought to crewmembers when and where they need it? Is the decision process changed by the communication medium? What relationships evolve over long durations between space and ground crews communicating through media?

Communication patterns have shown themselves to be highly sensitive indicators of crew functioning and have been used to measure coordination and productivity of the crew. The relevant importance of the communication/information area will vary according to the mission requirements. Research efforts should be initiated such that selected findings can be verified on Space Station Freedom and data are available to support the phase C/D milestone target dates of the two evolutionary case studies. The Mars Expedition case presents a timing problem since communication/information requirements impact vehicle planning. If an effort were to be undertaken to study this area of communication, useful information should be available about halfway through the C/D phase of the piloted vehicle.

5.1.2 Impact on/Applicability to Current Programs

The current research programs will provide the required life sciences human exploration information in all five critical areas: (1) advanced medical care, (2) artificial gravity countermeasures, (3) radiation protection, (4) life support, and (5) space human factors. Research centers are actively pursuing the development of computational tools for assessing crew performance in zero or partial gravity, and a collective effort by the NASA Centers has been undertaken to conduct a state-of-the-art review of human tolerance and performance in isolated, confined, and hazardous environments. Techniques to develop non-intrusive performance measures are also being developed.

The information derived from the life sciences artificial gravity program will be directly applicable to the development of biomedical countermeasures and will aid the entire countermeasure program. Current lunar outpost scenario plans assume the adequacy of lunar gravity in preventing detrimental human responses to microgravity. Should that not prove to be the case, plans for lunar surface outposts would have to be appropriately modified.

If gravitational "thresholds" are found to be different for different systems of the human body, this may influence the direction of countermeasure development and testing. It could also contribute to understanding basic biological principles regarding the importance of gravity in the evolution and function of these systems on Earth.

The major programmatic impacts for all five life sciences human exploration areas include manpower, facilities,

and new technologies. More staff is required, especially at the Field Centers, to fulfill the requirements for obtaining life sciences exploration information. In several life sciences areas, the same key personnel who form the core of the biomedical research program also form the core of the artificial gravity program. As exploration missions are planned, there will be an increased need to recruit experts in the near term to take on the responsibilities of key staff who will retire. It is critical to establish research fellowship/traineeship programs in each of the five life sciences areas to have an available human resource base for the exploration missions.

Ground- and flight-based facilities are needed to get the required life sciences information for exploration missions. Some facilities will require upgrading and minor modification (e.g., human-rated centrifuges, bed-rest facility), while other facilities will need to be developed (e.g., LifeSat, analog environments).

State-of-the-art technologies must be developed for the provision of advanced medical care, noninvasive behavioral analysis capabilities, life support subsystems, and biological dosimetry needed to assess current technologies and to identify the critical technology needs.

5.1.3 Support Required from Other NASA Programs

Human exploration missions will require support and interaction among all the NASA programs involved in life sciences and life support activities. A greater exchange of information as well as increased coordination and integration of programs and activities among all participants are required to support the development of critical life sciences information and technologies essential to meet the goal of human exploration of the solar system.

The Space Transportation System (STS) and Space Station Freedom will provide the base and test-beds to validate ground-based life sciences research findings. For the near term, continuing support from a planned series of STS/Spacelab missions and the Extended Duration Orbiter (EDO) program will provide information for the development of advanced medical care procedures and equipment for use on Space Station Freedom.

The STS is a key element in developing life sciences capability for exploration missions. Dedicated middeck accommodation on Shuttle and experimental access to the EDO will also be required to obtain critical life sciences information. The EDO will also provide the opportunity for testing biomechanical and performance changes in zero-g over time. The validation of the g-loading model can be tested using a modified Skylab treadmill during Shuttle and EDO missions. Early knowledge of the g "threshold," even by indirect means, would have an enormous impact on vehicle design and on planning long-duration Moon and Mars exploration missions, particularly if the gravitational threshold is above the equivalent of 0.4 Earth gravity. Continued support is needed from the Office of Space Flight, in terms of flight and experiment accessibility, for the purpose of updating and refining technologies in the life sciences areas, such as physiological loading and accurate dosimetry measurement of space radiation. The Office of Space Flight will also coordinate activities with the Life Sciences Division to provide launch capabilities for the planned reusable, recoverable free-flyer satellite (i.e., LifeSat). LifeSat will be required to fly in polar and geosynchronous orbits to obtain information for radiation dosimetry measurements.

Space Station Freedom will be a test-bed for studying many of the life sciences human exploration-related requirements. The Biomedical Monitoring and Countermeasures program is needed to develop countermeasures for long-duration space flight. Non-intrusive measures can be developed to obtain information on crew interactions, physical performance, quality of work, crew interaction with automated systems, work/rest schedules, and other habitability factors. On-orbit artificial gravity centrifuges will enable small animal and plant research, as well as the initiation of variable gravity studies. Regenerative life support systems for long duration exploration missions can be tested on Space Station Freedom. Improved EVA suits and capabilities can also be tested on Space Station Freedom.

Support will also be required from other Divisions within the Office of Space Science and Applications. Support from the Solar System Exploration Division will be needed to fly radiation detectors on planetary missions. This equipment is necessary to establish the experimental data from which tolerance levels, countermeasures, and warning systems for radiation can be developed. Some of the efforts of the Space Physics Division will also help to support the development of an accurate and reliable solar event protection and warning system for exploration missions.

5.1.4 Options and Trades

Life sciences requirements differ from those of, for example, propulsion or transportation systems, in that living systems, although flexible in many ways, are totally unaccommodating in others. For instance, if oxygen is lacking in the environment, oxygen must be supplied — there are no alternative solutions or workarounds. The trade options for meeting a requirement to supply oxygen fall in the category of design studies, not the life sciences evaluations. Therefore, the Life Sciences Division will participate in assessing trade options that are evaluated in the design process. Coordination with

the Office of Aeronautics and Space Technology and other NASA offices that examine technical alternatives to meet life sciences requirements will ensure evaluation of impacts to advanced medical care, artificial gravity/countermeasures, radiation, life support, and crew factors/human factors.

5.1.5 Conclusions

To undertake and successfully implement the human exploration of the Moon and Mars requires life sciences information in advanced medical care, countermeasures (with or without artificial gravity), radiation exposure, life support, and space human factors. The programs and activities planned by the Life Sciences Division conform to the direction and goals of NASA's exploration mission. Meeting the proposed schedules will require an aggressive program of life sciences activities conducted throughout NASA. This will require research and development activity over the present baseline program. For example, such an aggressive program is already contained in the combined OSSA Strategic Plan and the Human Performance element of Project Pathfinder. This program must continue to be implemented and integrated. In addition, an aggressive research and flight testing program will provide the life sciences data on which to base mission and vehicle design decisions. This program is essential and achievable; however, two factors impact the reliability of the data. The current milestone for making these decisions requires testing and validation on Space Station Freedom. Testing on Spacelab and other Shuttle missions will be useful and essential, but may be limited, due to both exposure duration and species constraints. Carefully controlled inflight experiments are needed to provide the required life sciences information to meet human exploration milestones.

In addition to the research and engineering requirements noted, the support of other NASA Headquarters organizations, particularly in providing early access to space flight testing, is required.

5.2 ROBOTIC MISSIONS

The robotic solar system exploration program is administered by the Office of Space Science and Applications (OSSA) and managed by OSSA's Solar System Exploration Division. The program uses a series of coordinated strategic missions to obtain scientific and technical information for research that will both advance scientific understanding and enable man to continue the exploration of the solar system. Many of these robotic missions will serve as precursors to human exploration missions by focusing the development of required technologies and demonstrating the engineering capabilities needed to conduct human exploration of the Moon and Mars.

5.2.1 The Solar System Exploration Division's Role in Human Exploration

During the last 2 fiscal years, the Solar System Exploration Division has been responsible for identifying programs to satisfy an emerging and maturing set of requirements identified and managed by OEXP. Responsibilities in support of human exploration are to define and execute robotic missions and studies to (1) establish and maintain the scientific and technical databases, and (2) provide selected technology and engineering demonstrations.

5.2.2 The Robotic Mission Definition Process

A first step in defining robotic missions to support human exploration of the solar system was to assess current OSSA Mars and lunar programs for their capability to satisfy known requirements. Candidate missions, either under development or being studied as part of the Solar System Exploration Division advanced studies effort, are the Mars Observer, Mars Rover/Sample Return, and Lunar Observer.

The Mars Observer is in development and on schedule for launch in September 1992. The Mars Observer will return valuable information compatible with any reasonable manned program; however, the mission will provide neither the high-resolution images necessary for support of manned operations nor the technology demonstrations deemed mandatory for future manned Mars operations. A supplemental Mars mission such as Mars Rover/Sample Return (MRSR) would be required.

The situation created by the expectation of human exploration of Mars in the near future is that crucial science and engineering data needed for designing manned systems and for conducting operations would be needed very early in the program. These data would be obtained by robotic missions. However, the time required for the design and conduct of a complex robotic mission (such as a sample return) is such that the precursor programs must begin soon. If they do not, the precursor missions could well become critical elements as well as the schedule drivers.

For human exploration of the Moon, the Lunar Observer, together with historical information gathered during Apollo, would be sufficient to satisfy existing precursor requirements. The Lunar Observer is planned to start in the early 1990s with flights anticipated in the late 1990s.

5.2.3 Approach to Meeting Exploration Requirements

The requirements of any robotic program designed to support a human landing must be structured to resolve

the following broad issues:

a. Landing site selection and certification

b. Crew safety issues and mission success concerns

c. Technology and engineering demonstrations

To resolve these issues, each mission might be characterized by a large, complex, long-duration flight that would satisfy several requirements, including high-resolution imaging, science investigations, and engineering and technology demonstrations. In addition, the emplacement of human exploration infrastructure elements that would support both the precursor and the manned program — e.g., communications orbiters, imaging orbiters, surface landing beacons, science surface packages, etc. — may ultimately be adopted as precursor objectives in a well-integrated human exploration/science program.

The Study Requirements Document (SRD) establishes requirements for unmanned robotic missions to (1) obtain science and engineering data needed to permit the design of safe and efficient spacecraft and crew systems, science and operations support equipment, and tools; (2) provide imaging and mapping information for selection of safe but scientifically interesting landing sites; and (3) demonstrate the engineering and technology that will be essential before human exploration can safely proceed.

A major U.S. commitment to a human Mars or lunar program would involve the equally important commitment to the robotic program. Without a national commitment to a Mars or lunar program, OSSA has reexamined and restructured its strategy to focus on the Moon and Mars, but not to abandon other elements of exploration objectives. The responsibility of OSSA in general and of the Solar System Exploration Division in particular includes the exploration of bodies other than Mars and the Moon. Thus, the OSSA strategic plan, while recognizing the importance of and growing interest in Mars and the Moon, reflects the objective of exploring the other inner and outer planets as well.

Barring additional commitment of resources or international participation, the allocation of OSSA resources to carry out a more intensive or aggressive Mars or lunar program would be accomplished only at the expense of other elements of the plan. Alternative program plans could be developed, which would result in a more flexible OSSA posture if a human exploration initiative were forthcoming.

To that end, two parallel efforts were initiated by the Solar System Exploration Division. The first effort was to identify candidate precursor programs that satisfy as many of the OEXP requirements as possible. The second effort was to ascertain the feasibility of integrating the program into the OSSA strategy consistent with established requirements and scheduling guidelines. MRSR was the prime program candidate for Mars, as Lunar Observer was for the Moon.

The Solar System Exploration Subcommittee of the NASA Advisory Council began a strategic planning effort before the FY 1991 budget submission to assess and restructure the solar system exploration program in a two-step process. The first step was to conduct a definition workshop in March 1989 to address program content, priorities, and alternative planning approaches. Timing was dictated by new administration, new budget, and further Soviet program insight. The second step was to conduct an evaluation workshop in June 1989 to refine the updated planning process and recommend near-term priorities before the FY 1991 budget submission.

Although the scope of the SSES workshops included the broader range of robotic exploration of the solar system — including the inner and outer planets as well as the comets and asteroids — much effort was focused on the exploration of Mars. The strategy of a "Mars centerpiece" was to create a flexible basic scenario that protected key options associated with the growing human exploration interest and the emerging potential for international participation.

Three scenarios were developed and are shown in table 5.2.3-I. The first, labeled "OSSA only," conforms to the current OSSA new start and funding guidelines as defined in the Office's 1989 Strategic Plan. The second scenario, labeled "Code E/Code Z," is based on the assumption that resources available to Code E (OSSA) to support the plan would be augmented by funds available to Code Z (OEXP) to make the program feasible. The third scenario, labeled "International," assumes that resources outside the OSSA strategic plan guidelines would be made available for conducting an international program of Mars exploration.

The final strategy recommended by the Solar System Exploration Subcommittee was to maintain a balanced exploration emphasis with programs such as Lunar Observer, Comet Nucleus Sample Return, and an outer planet orbiter and probe, while pressing for Mars surface exploration as soon as possible. The Mars surface probes would include a Mars International Network (and aeronomy), Mars Rovers, and Mars Sample Return.

5.2.4 Impact on/Applicability to Current Programs

The Lunar Observer Program is in the initial stages of the requirements definition process; the Mars Observer

TABLE 5.2.3-I.- MARS CENTERPIECE PROGRAM STRATEGY SUMMARY

New start year	OSSA-only scenario		Code E / Code Z scenario		International scenario	
	Launch year	Mission	Launch year	Mission	Launch year	Mission
					1994	Soviet Mars orbiter w/balloons, eggs and penetrators + U.S. VIMS experiment
1993 (moderate start)	1996	Mars imaging (super-Mars Observer camera) and aeronomy orbiter with penetrator network (4-6 sites)	1996	Mars imaging (super-Mars Observer camera) and aeronomy orbiter with penetrator network (4-6 sites)	1996	U.S. Mars super-Mars Observer camera and aeronomy orbiter with penetrator network (4-6 sites)
					*1998	Soviet Mars rovers with ESA communications relay and U.S. tracking/ops support
1996 (major start)	*2001	Mars limited autonomy rover	2001	Mars sample return with local rover	2001	U.S. Mars sample return with local rover
1999 (moderate start)			2005	Mars imaging (high-resolution)/ communications orbiter		
2001-02 (major start)	2005	Mars sample return with local rover	*2007+	Mars limited autonomy rover		
	*Rover would conduct a one-way traverse, deploying sample caches along the way at safe landing sites marked with beacons.		*Rover could be flown separately, or on an early piloted mission; teleoperated control either from Mars orbit or Earth.		*Rover would conduct a one-way traverse, deploying sample caches along the way at safe landing sites marked with beacons.	

is now in Phase C/D development and is scheduled for launch in September 1992. There is no impact to these programs due to precursor requirements.

The MRSR advanced study project is now completing Phase A studies. However, no significant impact is anticipated because the project has maintained a close liaison with the human exploration precursor requirements studies.

5.2.5 Conclusions

Mars is rapidly becoming unreachable within the near-term resources available to OSSA, primarily due to the divergence of available resources and the growing complexity of exploration requirements. Carrying the science investigation of Mars to more intensive levels will apparently require an evolving relationship with human exploration initiatives enhanced by the significant potential benefits of international cooperation.

5.3 TECHNOLOGY DEVELOPMENT

The Office of Aeronautics and Space Technology (OAST) is responsible for planning and implementing space research and technology (R&T) programs for future NASA mission programs. The development of the capabilities that would make the OEXP case study and robotic Mars precursor missions possible will depend upon successful research and technology development in a broad array of areas. This section discusses (1) the approach being used to determine and satisfy exploration technology needs, (2) specific technology needs identified during FY 1989 exploration studies, (3) the current content and status of technology programs supporting exploration, (4) advanced development options for exploration within NASA program offices, (5) outstanding strategic issues, and (6) summary conclusions.

5.3.1 Approach

The approach used to develop and implement plans for human and robotic solar system exploration technology involves several iterative steps. OAST analyzes exploration mission options, systems concepts, and identified technology needs, an analysis that includes active participation by user offices in technology assessments, options identification, and determination of appropriate levels of technology readiness and performance parameters. As a result of this analysis, potential technology needs and opportunities are identified, and ongoing technology programs are examined to assess the current state of the art in a technology area and to develop preliminary program plans where appropriate.

The information below provides first a restatement of exploration technology needs, including those for each of the FY 1989 human exploration case studies and for a robotic Mars precursor program, and then a status report on the NASA exploration technology program, focused on selected elements of the Civil Space Technology Initiative (CSTI) and the Pathfinder Program.

An overall planning framework (see figure 5.3.1-1) for exploration mission planning and exploration research and technology planning has been established by management in OAST, OEXP, and OSSA/Solar System Exploration Division. Within that framework, there is a planned negotiation process between flight system and R&T managers, keyed to the notion of increasing levels of "technology readiness" that are achieved within a mission-focused R&T program, such as Pathfinder, and demonstrated through specific technology breadboards and/or test-beds. Figure 5.3.1-2 details the established levels of technology readiness. Together, the mission/technology coordination schedule framework and the technology readiness levels provide a top-level context for specific exploration system technology needs identification, assessments, and prioritization.

As part of the FY 1989 OEXP study process, technology needs have been identified for each of the case studies, keyed in each instance to a specific mission system within the particular scenario. These technology needs were assessed and prioritized in terms of several criteria that were developed in parallel with criteria and planning processes used within the Pathfinder Program in FY 1988. These criteria included (a) *Commonality*: the degree to which the technology need is common to more than one case study, rather than unique; (b) *Timing*: how soon the technology must be brought to readiness to support a Phase C/D start for the specific mission system; and (c) *Risk/Challenge*: the inherent challenge in developing the technology, plus the uncertainty of that development given current and/or projected resources for that effort. Table 5.3.1-I summarizes the criteria applied to prioritize human technology needs.

5.3.2 Exploration Technology Needs and Assessment

Three case studies were examined during FY 1989: (1) Lunar Evolution, (2) Mars Evolution, and (3) Mars Expedition. Each mission involves an array of advanced technology needs, which either enable a particular mission design and its objectives or substantially enhance the mission design (through risk or cost reduction, or significant improvements in performance). Table 5.3.2-I summarizes the major technology needs for these case studies and provides an integrated assessment of criteria (including risk/challenge).

In addition to direct human exploration mission studies, OEXP, in combination with the Office of Space Science and Applications (OSSA) Solar System Exploration Division, conducted a robotic Mars precursor explora-

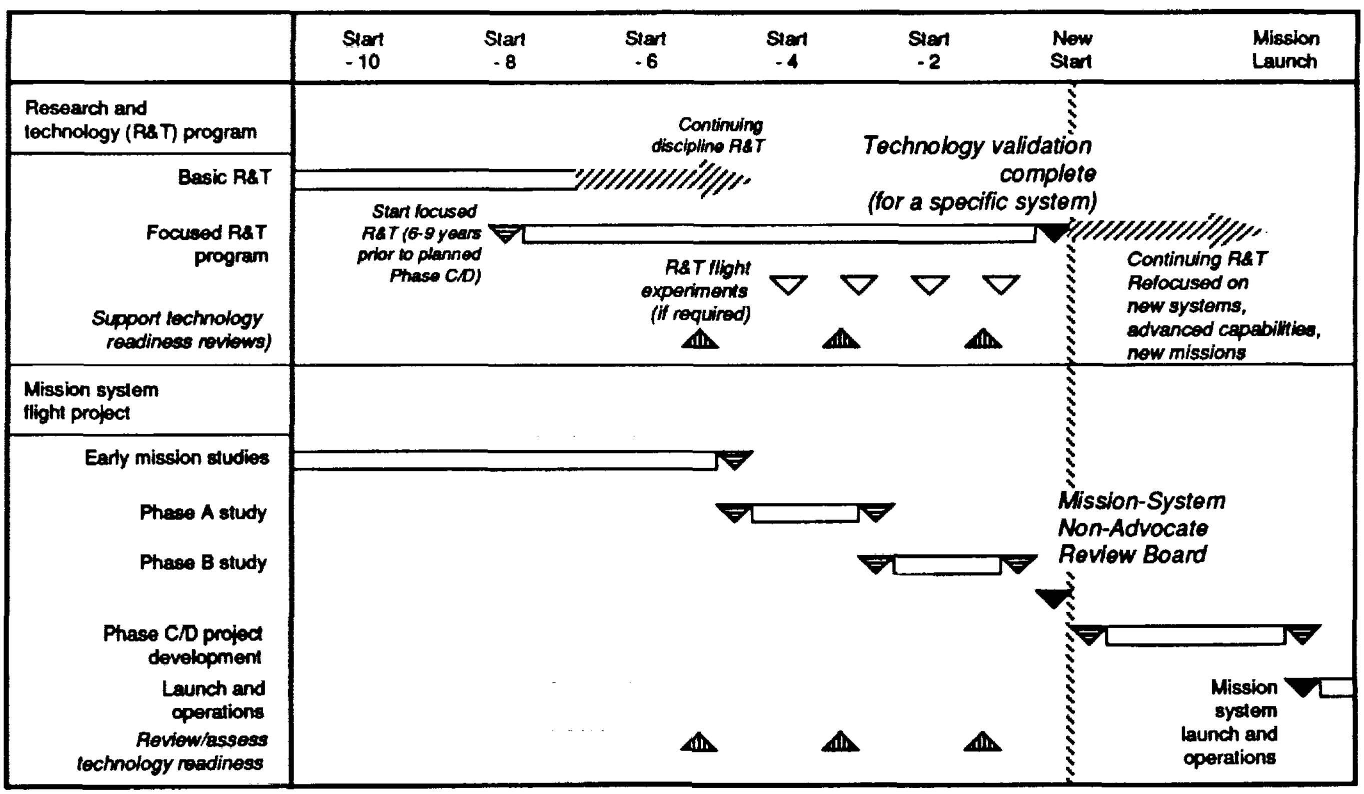


Figure 5.3.1-1.- Exploration mission and technology planning schedule.

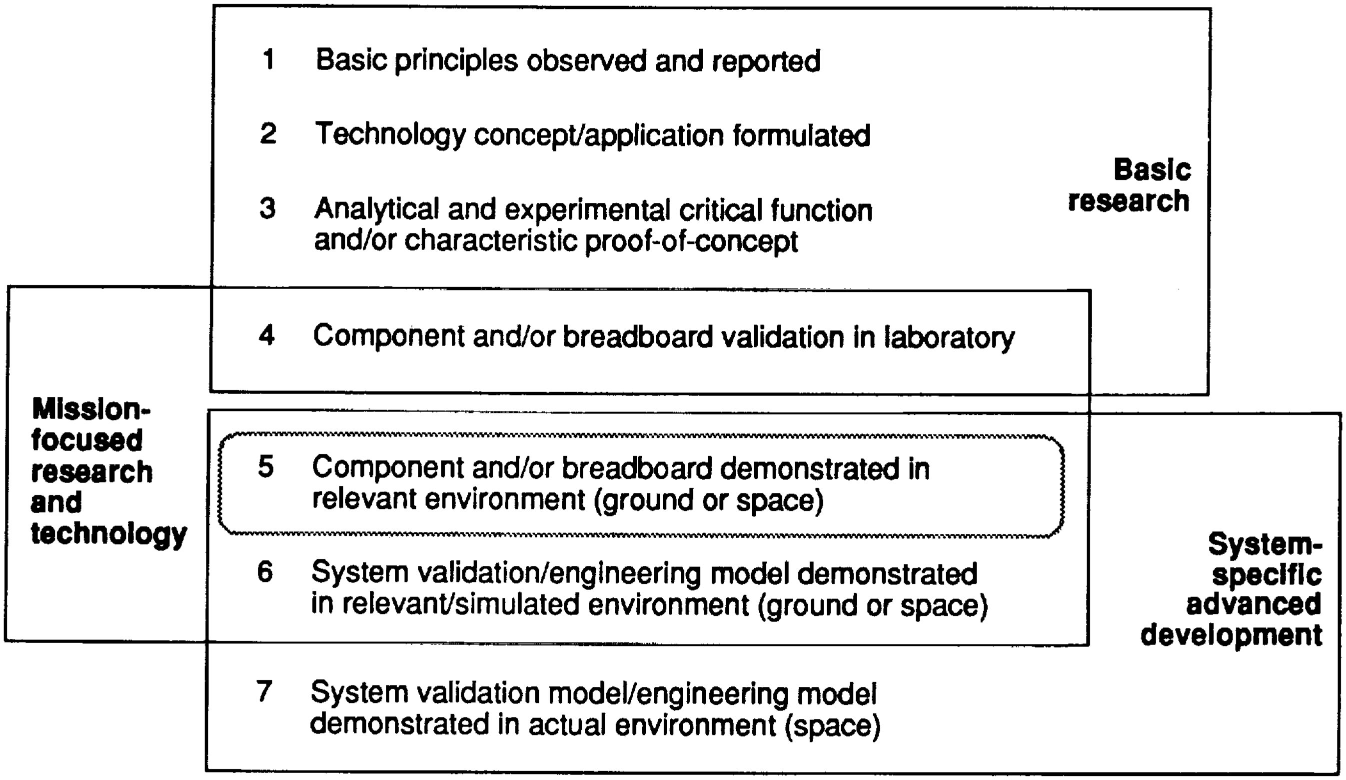


Figure 5.3.1-2.- Technology readiness levels and R&T program phases (including the approximate relationship between base and focused R&T programs).

TABLE 5.3.1-I.- EXPLORATION RESEARCH AND TECHNOLOGY RANKING CRITERIA

	System/Approach	Common	Unique
Need Categories	Enabling	I	II
	Enhancing	III	IV

	Period	Phase C/D	IOC
Needs Timing	A Near term	Post 1994	Pre 2004
	B Mid term	Post 1997	Pre 2007
	C Far term	Post 2000	Pre 2010

Development Risk/Challenge	1 High risk	Fundamental R&D and/or no program in place
	2 Med risk	Components and/or program in place with limited funding
	3 Low risk	On schedule;program fully funded

Definitions

Common: Required by all or most pathways and approaches. Specifically, a technology must be needed for both lunar and Mars scenarios in order to be in this category.

Unique: Required by only one or two pathways or approaches that NASA, as an agency, wishes to protect the option for implementing.

Enabling: Those technologies which must be available in order for the mission to be a success either from a technical feasibility/performance aspect or from an affordability aspect.

Enhancing: Those technologies that yield a significant net positive benefit in terms of capability and/or affordability.

TABLE 5.3.2-I.- TECHNOLOGY NEEDS BY RANK

Technology	Ranking		Timing	Risk	Integration area
Construction technology	Enabling	Common	Near	Med	Planetary surface systems
Surface transportation technology	Enabling	Common	Near	Med	Planetary surface systems
Regenerable life support system	Enabling	Common	Near	Med	Planetary surface systems
Trace contaminant control	Enabling	Common	Near	Med	Planetary surface systems
Waste management	Enabling	Common	Near	Med	Planetary surface systems
Water recovery/management	Enabling	Common	Near	Med	Planetary surface systems
In-space vehicle processing/servicing	Enabling	Common	Near	Med	Orbital node
Aerocapture (low energy @ Earth)	Enabling	Common	Near	Med	Transportation
Radiation protection	Enabling	Common	Near	Med	Transportation
Surface power (<1 MWe)	Enabling	Common	Near	Low	Planetary surface systems
EVA systems technology	Enabling	Common	Near	Low	Planetary surface systems
Atmosphere revitalization	Enabling	Common	Near	Low	Planetary surface systems
Cryogenic fluid supply/storage/management	Enabling	Common	Near	Low	Orbital node
Cryogenic fluid transfer/handling	Enabling	Common	Near	Low	Orbital node
Chemical ascent/descent engine	Enabling	Common	Near	Low	Transportation
Cryogenic fluid supply/storage/management	Enabling	Common	Near	Low	Transportation
Cryogenic fluid transfer	Enabling	Common	Near	Low	Transportation
Long-lived life support units	Enabling	Common	Near	Low	Transportation
Advanced chemical transfer engines	Enabling	Common	Near	Low	Transportation
In-space assembly - vehicle level	Enabling	Unique	Near	Med	Orbital node
Aerocapture (low energy @ Mars)	Enabling	Unique	Near	Med	Transportation
Aerocapture entry/landing @ Mars	Enabling	Unique	Near	Med	Transportation
Nuclear thermal rocket propulsion	Enabling	Unique	Near	Low	Transportation
Autonomous rendezvous and docking	Enhancing	Common	Near	High	Orbital node
Mobile power systems	Enhancing	Common	Near	Med	Planetary surface systems
Thermal control	Enhancing	Common	Near	Med	Planetary surface systems
In-space assembly - element level	Enhancing	Common	Near	Med	Orbital node
Autonomous landing	Enhancing	Common	Near	Med	Transportation
Dust contamination control	Enhancing	Common	Near	Low	Planetary surface systems
Ka-band communications technology	Enhancing	Common	Near	Low	Transportation
Lunar oxygen production	Enabling	Unique	Mid	Med	Planetary surface systems
Mining technology	Enabling	Unique	Mid	Med	Planetary surface systems
Mars water extraction	Enabling	Unique	Mid	Med	Planetary surface systems
Aerocapture (high energy @ Earth)	Enhancing	Unique	Near	Med	Transportation
Artificial gravity vehicle	Enhancing	Unique	Near	Med	Transportation
Direct entry @ Earth (high energy)	Enhancing	Unique	Near	Med	Transportation
Artificial-g vehicle deployment and control	Enhancing	Unique	Near	Low	Transportation
Tethers	Enhancing	Unique	Near	Low	Transportation
Parachute system (Earth/Mars)	Enhancing	Unique	Near	Low	Transportation
Inflatable structures	Enhancing	Common	Mid	Med	Planetary surface systems
Propellant storage and transfer	Enhancing	Common	Mid	Med	Planetary surface systems
Mineral beneficiation	Enhancing	Common	Mid	Low	Planetary surface systems
Surface power (>1 MWe)	Enabling	Common	Far	Med	Planetary surface systems
Food production	Enabling	Common	Far	Med	Planetary surface systems

TABLE 5.3.2-I.- (CONCLUDED)

Technology	Ranking		Timing	Risk	Integration area
Essential element extraction	Enhancing	Common	Far	Med	Planetary surface systems
High power electric propulsion (MW class)	Enhancing	Common	Far	Med	Transportation
Nuclear power for NEP	Enhancing	Common	Far	Med	Transportation
Aerocapture (dual use @ Mars/Earth)	Enhancing	Unique	Far	High	Transportation
Lunar ceramics production	Enhancing	Unique	Far	Med	Planetary surface systems
Lunar hydrogen production	Enhancing	Unique	Far	Med	Planetary surface systems
Lunar metals production	Enhancing	Unique	Far	Med	Planetary surface systems
Mars atmospheric oxygen extraction	Enhancing	Unique	Far	Med	Planetary surface systems
Phobos/Deimos water extraction	Enhancing	Unique	Far	Med	Planetary surface systems
In situ propellant engines	Enhancing	Unique	Far	Med	Transportation
Solar power for SEP (MW class)	Enhancing	Unique	Far	Med	Transportation

tion program study. The Mars Rover/Sample Return (MRSR) mission study defined a set of technology requirements that will enable or substantially enhance any robotic Mars precursor exploration program. Table 5.3.2-II summarizes the major technology needs for these robotic precursor missions and provides an integrated assessment of criteria (including risk/challenge) and current accommodation within the projected FY 1990 OAST research and technology programs. [Fiscal year 1990 (and runout) budget data were used in developing the assessment of overall R&T program uncertainties (risks) provided here.]

5.3.3 Research and Technology Programs

Two OAST programs provide the primary R&T support for human exploration mission planning. First, the Civil Space Technology Initiative (CSTI), begun in FY 1988 to fill existing gaps in the R&T base, includes several element programs that are applicable to exploration needs. Second, following extensive coordination with the Sally Ride Space Leadership Planning Group activity, OAST initiated the Pathfinder Program in FY 1989 to develop needed technologies for future human and robotic solar system exploration missions.

5.3.3.1 Civil Space Technology Initiative (CSTI)

OAST's CSTI program is a major focused technology effort directed toward developing capabilities to support access to, operations in, and science in, Earth orbit. Within these three thrusts, CSTI is composed of some 10 element programs, of which two have particular targeted applications within human exploration missions.

Aeroassist Flight Experiment. The Aeroassist Flight Experiment (AFE), one of the elements of the CSTI transportation thrust, will investigate the critical vehicle design technologies and upper atmospheric characteristics applicable to future low-energy aeroassisted space transfer vehicles. Managed by MSFC, the AFE Program focuses on a single major flight experiment, including a carrier vehicle, a blunted, rake-cone aeroshell using Shuttle-type thermal protection system tile technology and incorporating a number of experiments such as radiative heating and wall catalysis measurements, forebody aerothermal characterization, and wake flow base heating measurements.

High Capacity Power. LeRC is developing the technology to provide long-duration, high-capacity power by dramatically improving the thermal-to-electric conversion efficiency. The program primarily addresses development of advanced free-piston Stirling engine technology, but it also includes improved efficiency thermoelectric converters. Both of these efforts will provide a technology base that could be used to provide substantial improvements in SP-100 efficiency for lunar outpost or other exploration mission applications. As funding permits, the program will also develop advanced waste heat radiator concepts to improve the specific mass characteristics of currently planned SP-100 radiators.

Other CSTI Programs. Three other elements in CSTI are providing varying degrees of technology support for future mission applications. These elements include: (1) the robotics program, (2) the autonomous systems program, and (3) the controls-structures interaction program. Each of these efforts is contributing materially to a technology base upon which more focused future exploration R&T programs will build.

5.3.3.2 Pathfinder Program

The Pathfinder Program will provide technologies for solar system exploration mission applications, includ-

TABLE 5.3.2-II.- TECHNOLOGY NEEDS BY RANK FOR ROBOTIC MARS PRECURSORS

Technology	Ranking	Timing	Risk	System
Surface penetrators/probes	Enabling	Near	High	
Surface transportation technology	Enabling	Near	Med	Demonstration
Aerocapture (low energy)	Enabling	Near	Med	Demonstration
Surface power (<1 MWe)	Enabling	Near	Med	
In situ science systems	Enabling	Near	Med	Demonstration
Mobile power systems	Enabling	Near	Med	
Autonomous landing	Enabling	Near	Med	Demonstration
Autonomous rendezvous and docking	Enabling	Near	Low	Demonstration
Radiation protection	Enhancing	Near	Med	
Dust contamination control	Enhancing	Near	Med	Demonstration
Information management	Enhancing	Near	Med	Demonstration
Ka-band communications technology	Enhancing	Near	Med	Demonstration
Chemical ascent/descent engine	Enhancing	Near	Low	
Aerocapture (high energy)	Enhancing	Mid	High	Demonstration
In situ resource utilization	Enhancing	Mid	Med	Demonstration
Nuclear thermal rocket propulsion	Enhancing	Far	High	Demonstration
In-space vehicle processing/servicing	Enhancing	Far	Med	Orbital node
Nuclear surface power	Enhancing	Far	Med	Demonstration
Cryogenic fluid management	Enhancing	Far	Med	Orbital node
High power electric propulsion (MW class)	Enhancing	Far	Med	Demonstration
Nuclear power for NEP	Enhancing	Far	Med	Demonstration
Solar power for SEP (MW class)	Enhancing	Far	Med	Demonstration

Demonstration listings represent potential applications of advanced technology within a precursor mission that could serve as a demonstration of that capability for later Humans-to-Mars applications

ing both robotic spacecraft systems and piloted missions to the Moon and Mars. Pathfinder addresses exploration technology needs within four broad areas: (1) surface exploration, (2) in-space operations, (3) humans in space, and (4) space transfer.

1. Surface Exploration: The surface exploration program area is developing critical technologies to enable or significantly enhance future piloted and robotic exploration of planetary surfaces. Research and technology have been started in four element programs: (1) planetary rover; (2) sample acquisition, analysis, and preservation; (3) autonomous lander; and (4) surface power.

Planetary Rover. The planetary rover program is developing the technology required to enable semi-automated and piloted exploration of extensive areas of the Moon and Mars. Particular progress has been achieved in the area of advancing on-board semiautonomous navigation, and in sophisticated, legged mobility systems. Program objectives include development of both wheeled and legged mobility systems, with semiautonomous onboard navigation, and the capability to traverse either obstacles or surface irregularities between 1 and 2 meters in size.

Sample Acquisition, Analysis, and Preservation. This program, being implemented by researchers at JPL as well as at JSC and other NASA Centers, is developing specialized tools for sample acquisition and advanced sample analysis sensors, such as acousto-optical tunable filters. The program will develop the capability to obtain a suite of targeted atmospheric, surface regolith, rock coring, and subsurface regolith samples, and to preserve these samples in acceptable pressure and temperature conditions for return to Earth for more detailed analysis.

Autonomous Lander. JSC and JPL are also developing strategies and trajectory algorithms, guidance, navigation, and control technologies, and software and sensors for adaptive hazard avoidance during the final moments of automated landing on planetary surfaces. The program is developing capabilities to support both piloted and robotic Mars precursor landers. Characteristic technology performance objectives for landing accuracy range down from ±10 kilometers from the target site for an initial landing, to ±50 to 100 meters from the target site for landing near previously landed spacecraft, with

the capability to detect and avoid surface hazards of ≥ 1 meter in size, and a probability of safe landing >96 percent.

Surface Power. LeRC is developing essential technologies for advanced solar-based power systems, including highly efficient, low-mass regenerative fuel cells (RFCs), and low-mass solar photovoltaic (PV) arrays designed to operate on a planetary surface. Goals and objectives include the development of amorphous silicon PV cells for use in arrays with a specific power of approximately 300 watts/kilogram, regenerative fuel cells with an energy density of 1,000 watt-hours/kilogram for lunar applications and 500 watt-hours/kilogram for Mars applications, and advanced electric power management systems with specific mass performance of approximately 55 kilograms/kilowatt. The overall surface power system technology performance objectives are for 3 watts/kilogram for lunar applications and 8 watts/kilogram for Mars applications.

Other Programs. An additional program, photonics, has been planned; however, this program has been deferred to FY 1991. When this program is initiated, it will address the development of selected photonics-based sensor and data processing components (such as spatial light modulators), and subsystems (such as image correlators) to make supercomputer-class processing applications feasible within a variety of exploration mission systems, such as rovers and landers.

2. In-Space Operations: The Pathfinder in-space operations thrust is directed at the development of key technologies that will enable a wide variety of high-leverage operational capabilities for solar system exploration. Within this area, research and technology are underway in the elements of: autonomous rendezvous and docking, in-space assembly and construction, cryogenic fluid depot, and space nuclear power (the NASA contribution to the ongoing SP-100 Ground Engineering System technology project).

Autonomous Rendezvous and Docking. JSC, working with MSFC and JPL, will develop and test a variety of sensors, including laser ranging and vision, techniques for sensor fusion and responsive data processing (such as neural network architectures), and adaptive docking mechanisms. Goals include system reliability in excess of 99.99 percent, with a range resolution of ≤ 1 centimeter, over a range from 0 to 1.0 kilometer, and 1 percent of the distance over a range of 1.0 to 100 kilometers.

In-Space Assembly and Construction. LaRC is developing and will demonstrate capabilities for precise manipulation of large, massive space systems and permanent joining of lightweight, high-strength space structures. Program performance objectives include development of low-mass permanent mechanical joints for 22.7, 45.4, and 113.4 t loads, "space crane" technologies to support a reach of approximately 100 meters, and a load capability of 22.7 to 226.8 t.

Cryogenic Fluid Depot. LeRC is developing critical technologies for cryogenic propellant storage, transfer, and management, as well as large-scale cryogenic containment and refrigeration components and systems. Key technology issues include minimizing long-term cryogenic fluid loss (e.g., through boiloff or transfer losses), providing critical instrumentation (e.g., to determine remaining cryogen levels in zero gravity), and creating integrated analytic design tools to support future mission cryogenic systems development programs.

Space Nuclear Power. The SP-100 program is a joint NASA, Department of Defense (DoD), and Department of Energy (DoE) program, managed by the DoE, that will develop both space reactor technology and selected subsystem technologies, such as thermoelectric conversion and thermal management. The Space Nuclear Power Program, managed by JPL and the Los Alamos National Laboratory, will provide the capability to design and develop space power systems for both in-space and planetary surface applications. Performance objectives include power generation in the 10 to 1,000 kWe range, lifetimes of at least 7 years, and operations in both zero-gravity and planetary surface applications.

Other Programs. Two additional programs have been planned, but deferred until FY 1991: resource processing pilot plant and optical communications. The resource processing pilot plant will develop a diverse assortment of technologies, including extraction and processes and materials collection and handling. The optical communications program will develop a flight experiment package to demonstrate laser communications from a deep space platform, such as the Cassini spacecraft planned for robotic Saturn exploration.

3. Human Performance: The objective of the Human Performance program area is to conduct research to define requirements and develop technologies that will enable or enhance long-duration human space missions. Technology efforts have started in three critical elements: (1) extravehicular activity suit, (2) space human factors, and (3) physical-chemical life support.

Extravehicular Activity (EVA) Suit. The Ames Research Center (ARC) and JSC are developing for future human planetary and lunar surface operations portable life support systems, suit concepts, gloves and end-effector technologies, EVA tools and mobility systems, and suit information and control interfaces. A suit component breadboard has been developed at ARC for application in LEO, which will form the basis for development of

planetary surface EVA systems. Key system-level EVA technology objectives include the duration and number of EVAs to be performed per day, suit internal pressure and carbon dioxide partial pressure (and disposal), and mobility requirements.

Space Human Factors. JSC and ARC are also developing advanced human-automation-robotic systems and evaluating crew operations support and capability enhancement systems. Program objectives include development of advanced workstations, design and breadboard testing of mission-focused human interactive systems, and assessments of proposed planetary surface habitats.

Physical-Chemical Life Support. ARC, JSC, and MSFC are developing technologies for long-lived, highly reliable life support systems. Program objectives include air revitalization, water reclamation, waste treatment, and air and water quality control technologies. Critical program objectives will revolve around the degree of closure of each major life support cycle (air, water), the system efficiency, mean time between failures, requirements for maintenance and servicing, and the use of in situ resources to supplement Earth-supplied logistics.

Requirements are also being defined in two other programs: bioregenerative life support requirements and human performance requirements. These two Pathfinder programs are being managed at NASA Headquarters by the Office of Space Science and Applications (OSSA) Life Sciences Division.

Bioregenerative Life Support Requirements. This program is defining feasibility and engineering and technology requirements for bioregenerative life support system components; the scope of the program ranges from requirements for food supplementation modules through integrated controlled ecological life support systems (CELSS) for specific mission applications. In particular, the program will define requirements (both engineering and technology) for food supplementation, as well as mission-specific technology requirements to support CELSS for future long-duration human exploration missions.

Human Performance Requirements. This activity is determining detailed human requirements from top-level exploration mission objectives in the areas of extravehicular activity, space human factors, radiation effects and countermeasures, and artificial gravity requirements. Through work at ARC and JSC, the following objectives are being pursued: (1) definition of a quantitative model relating g-loading to physiological effects, (2) recommendations for radiation protection countermeasures in space, (3) habitability and design requirements, and (4) physiological and environmental requirements for advanced extravehicular activity systems.

Other Programs. The crew protective systems technology program has been deferred; when it begins, it will address technologies for radiation protection and artificial gravity systems for long-duration human exploration missions.

4. Space Transfer: The Pathfinder space transfer program area will develop advanced space transportation technologies that will improve the effectiveness of future exploration vehicle systems. In the area of technology to improve space transfer systems, two Pathfinder element programs have been initiated: chemical transfer propulsion and high-energy aerobraking.

Chemical Transfer Propulsion. This program, managed by LeRC, will develop and demonstrate the critical technologies for an advanced expander cycle liquid oxygen (LOX), liquid hydrogen chemical transfer engine. Development methodologies, engine systems, and mission-focused advanced engine components, including health monitoring and control harness equipment, will be demonstrated in one or more breadboard cryogenic engines, with targeted performance parameters including thrusts between 2.3 and 22.7 t, throttling capability of 20:1, man-rating and space-basing capability, and a vacuum-rated specific impulse (Isp) of approximately 490 seconds. Component and sub-component research and testing during the 1970s and early 1980s are the foundation upon which the Pathfinder chemical transfer propulsion program's advanced engine technology development is built.

High-Energy Aerobraking. ARC, JSC, LaRC, and JPL are developing concepts and component technologies for future space vehicle aeroassist systems. Program objectives include development of Mars-specific aerothermodynamic computation fluid dynamic codes, reusable thermal protection system materials, and adaptive, onboard guidance, navigation, and control techniques. Program goals include the capability for aerobraking at Mars in a range from 6 to 10 kilometers/second, and at Earth in a range from 11.5 to 14 kilometers/second, with entry g-loads $\leq$5 g's, and a total aerobrake thermal protection system mass $\leq$15 percent of the total vehicle dry mass.

Other Programs. An element to develop cargo vehicle propulsion has been planned, but deferred to fiscal year 1991. When initiated, this program will develop and demonstrate the technologies needed to enable highly efficient interplanetary electric propulsion capabilities, such as ion thrusters for the 10 to 100 kWe range, and magnetoplasmadynamic thrusters in the multi-megawatt power range.

5.3.3.3 Summary Assessment

Overall, the Pathfinder Program and selected elements of the Civil Space Technology Initiative represent an excellent foundation of focused research and technology support for future solar system exploration mission options. Several high-leverage or enabling Pathfinder R&T programs have been deferred due to funding limitations. These include (1) the resource processing pilot plant (which will develop the technologies needed to enable lunar oxygen production), (2) crew protective systems technology (which will develop radiation protection and artificial gravity technologies), and (3) cargo vehicle propulsion (which will develop and demonstrate key technologies for either solar electric propulsion systems or nuclear electric propulsion).

In addition to planned but deferred R&T programs, key technology areas that are not covered in the FY 1990 focused R&T programs include: (1) surface habitats and construction, (2) planetary surface penetrators, (3) bioregenerative life support technology, and (4) surface thermal management systems. Moreover, several high-leverage technologies are not covered in the current program; for example, nuclear thermal rocket (NTR) technology. Table 5.3.3-I provides a summary assessment of technology programs versus exploration mission technology needs.

5.3.4 Advanced Development Program Options

In general, focused research and technology efforts such as Pathfinder or selected elements of CSTI that are directed toward meeting exploration mission needs will develop a continuing suite of technology options for potential application to exploration mission systems.

TABLE 5.3.3-I.- SUMMARY ASSESSMENT OF SUPPORT TO EXPLORATION MISSION APPLICATIONS PROVIDED BY PATHFINDER AND SELECTED CSTI ELEMENT PROGRAMS

Technology program coverage assessment	Technology program assessment			Status
	Pathfinder	CSTI	Comments	
Surface transportation	Planetary rover	Robotics, AI	—	Good
In situ science systems	Sample acquisition, analysis, and preservation	Robotics, AI	—	Good
Autonomous landing	Autonomous lander	—	—	Good
Surface power systems	Surface power	High capacity power	—	Good
Surface penetrators/probes	—	—	Program needed	Poor
Autonomous rendezvous & docking	Autonomous R&D program	—	—	Good
Vehicle processing	—	Robotics, AI	Needs focused program	Poor
In-space construction	In-space assembly and construction	Robotics	Pathfinder program deferred	Poor
Cryogenic fluid management	Cryogenic fluid depot	—	COLDSAT experiment	Good
In situ resource utilization	—	—	Pathfinder program deferred	Poor
Information/communications	—	Data high rate/capacity	Pathfinder program deferred	Poor
Surface thermal management	—	High capacity power	Needs focused program	Poor
Surface construction	—	Robotics	Needs focused program	Poor
Surface habitat systems	—	—	Program needed	Poor
Life support systems	Physical/chemical and bioregenerative life support systems	—	—	Good
EVA systems	EVA/suit	—	—	Good
Radiation protection	High performance requirements	—	Pathfinder R&T deferred	Poor
Artificial gravity	High performance requirements	—	Pathfinder R&T deferred	Poor
Dust contamination control	EVA/suit	—	More emphasis needed	Poor
Nuclear propulsion (NTR/NEP)	SP-100	High capacity power	Pathfinder program deferred	Moderate
Nuclear surface power	SP-100	High capacity power	—	Good
Cryogenic ascent/descent engine	Chemical transfer propellant	—	—	Good
Aerobraking (low energy)	High energy aerobraking	AFE	—	Good
Aerobraking (high energy)	High energy aerobraking	—	—	Good
Advanced chemical engine	Chemical transfer propellant	—	Program under study	Moderate
	Pathfinder	CSTI	Comments	

Exploration advanced development programs, on the other hand, will provide focused development of specific — and critical — reference design technologies and/or subsystems, beginning approximately with the initiation of Phase B for a specific system. Together, focused R&T programs managed by OAST and advanced development programs for specific systems managed by the appropriate user offices are being planned to provide the technologies that will be essential to the success of any future exploration program. Selected prospects for advanced development and pre-project testing include (1) low-Earth orbit vehicle processing and fueling, including deployment and/or assembly of space transfer vehicle aeroshell systems; (2) life-testing of prototype systems, such as life support, avionics, and specific engines; and (3) large-scale ground-based simulated environmental testing of prototype lunar outpost operational equipment.

5.3.4.1 Prototype Development and Life Testing

Exploration missions will be required to provide levels of system performance that are one or more orders of magnitude longer in duration, and with higher reliability, than current Shuttle systems or historical Apollo systems. A considerable degree of the required technology readiness to provide necessary levels of project management confidence in systems performance will be provided by the research and technology program. However, in many cases, critical specific systems or subsystems may require prototype development and testing during pre-project advanced development activities. In particular, the following systems may require prototype development and life testing as part of exploration advanced development programs: systems for physical-chemical and bioregenerative life support, extravehicular activity systems, in-space and surface power, and advanced engine systems (including life testing of both cryogenic propulsion systems and electric propulsion systems).

5.3.4.2 Simulated Environmental Testing

A wide assortment of new and innovative aerospace systems are being considered for application within human lunar and Mars missions. Key systems that may require extended simulated environmental testing include EVA systems, surface power systems, construction and mining equipment, and advanced surface habitat systems (such as inflatable or otherwise deployable large volume modules and erectable regenerative life support chambers).

5.3.4.3 LEO Operations Demonstrations

Finally, a variety of STVs are being considered for exploration mission applications, including both lunar and Mars transfer vehicles. Many, if not all, of these vehicles will require a variety of supporting systems in low-Earth orbit (LEO) on or near Space Station Freedom. These vehicles and their supporting LEO systems will require substantial advanced development programs prior to the initiation of detailed design, development, test, and evaluation. Specifically, critical capabilities for exploration systems include on-orbit systems check-out and testing for cryogenic fluid transfer and storage, on-orbit assembly and/or maintenance and adjustment of structural elements, and proximity operations (such as automated rendezvous and docking). Each of these may require advanced development activities and demonstrations in LEO to verify key performance characteristics prior to mission application.

5.3.5 Strategic Issues and Concerns

Technology Transfer. There is a continuing need for specific and vigorous management systems to ensure that effective technology transfer occurs from OAST technology programs into human exploration flight projects. Potential management systems include: (1) a series of "technology review boards" coordinated with the phases of mission design could provide a forum for technology selection and readiness certification that could substantially improve the effectiveness of advanced technology transfer (such review boards would be chartered by the responsible flight project's line management), with substantial participation by non-advocate technologists; and (2) a coordinated strategy between the OAST space research and technology programs and mission program office advanced development investments that is being formulated to help ensure transfer between these efforts.

Technology Roles. There is a need to address questions of technology content and roles of projected international partnerships in solar system exploration missions beyond the Moon. In particular, the a priori technology content of the mission responsibilities of international partners in future mission options should not automatically relegate the advanced technology in the program to a single partner. These questions should be considered by the appropriate managers in strategic and programmatic planning.

Technology and Cost Estimation. The importance of cost estimation in the mission design process cannot be understated. Moreover, there is a direct relationship in current cost modeling approaches between technical complexity and "inheritance" that bears directly upon the questions of technology readiness level goals for a supporting technology program. An assessment is needed of potential approaches to coordinate research and development technology readiness and mission cost estimation processes within OAST and the appropriate user program office for the solar system exploration

technology and mission programs.

Robotic Exploration. The NASA Office of Space Science and Applications (OSSA) Solar System Exploration Division continues to formulate strategic options and preliminary implementation program scenarios for future U.S. robotic solar system exploration missions. There is a strong potential role for the OSSA/Solar System Exploration Division robotic exploration program in technology demonstrations and capability development for later human mission applications, as well as an existing responsibility with OAST for the direct development of technology for those missions. Integrated assessments will be conducted of the coordinated OAST and OSSA/Solar System Exploration Division program roles in research, development, and demonstrations in support of human exploration. Moreover, the technologies developed for the human exploration program (including the robotic Mars precursor program) will provide a foundation of technology for an exciting array of other robotic exploration missions, including future outer planet missions. An advanced Jovian system explorer, for example, could utilize space nuclear power, interplanetary electric propulsion, autonomous landing systems, and in situ sampling systems developed through the human exploration program.

5.3.6 Summary and Future Directions

Most technologies needed to enable or substantially improve the engineering and/or cost performance of projected exploration mission options are covered by the Pathfinder Program and selected elements of the Civil Space Technology Initiative. However, several specific technology areas are not currently being covered by the FY 1990 technology program. For example, these include surface construction and surface habitats. Areas not currently being worked also include planned but deferred element programs, such as the resource processing pilot plant program and the crew protective systems technology program (for radiation protection). These technology areas and others will be examined in more depth as part of planning for future exploration R&T.

SECTION 6

Options, Alternatives, and Trades

In addition to the systems definition studies performed by the Integration Agents (IAs) in direct response to specific focused case study requirements, the OEXP activities also include studies and product development in three identifiable categories: (1) special assessments, (2) controlled trade studies, and (3) emerging case studies.

1. Special Assessments: Special assessments focus on "high leverage" issues that are independent of specific case studies. The assessments generally cover a broad subject area with potential for significant benefit to all mission options. Four special assessments — (1) power systems, (2) propulsion systems, (3) life support systems, and (4) automation and robotics — are discussed in sections 6.1, 6.2, 6.3, and 6.4 respectively.

2. Controlled Trade Studies: Controlled trade studies are those that affect more than one case study, are case-study independent, or involve multiple integration areas. The studies represent parametric analyses across a broad range of options, and the results are essential to further mature the case studies or enable a technical "assault" on the case study constraints. Three trades — Earth-Moon node location, lunar liquid oxygen leverage, and launch/on-orbit operations — are discussed in sections 6.5, 6.6, and 6.7 respectively.

3. Emerging Case Studies: Emerging case studies are candidate focused case studies that are not yet sufficiently mature for release from the MASE analysis process. Upon further analysis by MASE and review by OEXP, an emerging case study may be adopted as a future focused case study, or it may be dropped from consideration. Two emerging case studies, the Lunar Oasis and a Near-Earth Asteroid Expedition, are discussed in sections 6.8 and 6.9 respectively.

6.1 POWER SYSTEM SPECIAL ASSESSMENT

This section discusses the power system alternatives that were reviewed for spacecraft and lunar and Mars surface applications, building on the assessments conducted in FY 1988. Spacecraft power options were assessed for nuclear electric and solar electric cargo vehicles. A concept was investigated for quick deployment of a nuclear power plant for use on the lunar surface. Preliminary requirements were assessed for Mars stationary surface power systems, and power system alternatives for a wide range of surface mobility requirements were reviewed.

6.1.1 Spacecraft Power

6.1.1.1 Spacecraft Nuclear Power for Electric Propulsion

SP-100 Reactor Scalability. The only U.S. space nuclear reactor development program is the SP-100, a joint DoD, DoE, and NASA program with the goal of providing a nuclear power system that weighs less than 30 kg/kWe and has a full power lifetime of 7 years. The first reactor being developed in the SP-100 program (the ground engineering system) is a 2.5 MWt reactor with thermoelectric conversion that produces 100 kWe.

Studies performed by General Electric at the direction of the Lewis Research Center show ways in which an SP-100 type reactor can be up-scaled to a thermal power range of 10 to 50 MWt without increasing the fuel, materials, and component technologies currently being developed in the SP-100 ground engineering system program. However, additional testing may be required to qualify sliding reflectors, scaled-up pumps and motor drives, increased flow rates, Li_7H shielding, and extended-life fuel and materials.

The SP-100 reactor technology appears to be flexible enough to meet a wide range of mission applications. In addition to high-power NEP missions, these SP-100 derivative reactors may be used for planetary surface mining and manufacturing.

NEP Power System Options Assessment. A chemical propulsion system augmented by aerobraking is the baseline propulsion system for the Lunar and Mars Evolution case studies. A NEP system is currently being considered as a technology enhancement for both studies. A special assessment addressing the characteristics of a NEP cargo vehicle as applied only to the Mars Evolution case study showed that, sufficiently shielded, NEP-driven vehicles could be used for piloted interplanetary transfers between high-Earth orbit and Mars with trip times comparable to chemical propulsion system alternatives. In this application, NEP could reduce the dependence of transportation systems on in situ propellants.

The power system of a NEP vehicle consists of the reactor heat source, power conversion, heat rejection, and power conditioning. Several candidate nuclear power generation concepts, including closed Brayton, free-piston Stirling, potassium Rankine, and in-core thermionic cycles, were compared on the basis of mass, radiator area, and technical maturity. The dynamic systems would be integrated with an SP-100-type reactor through a heat exchanger. Shielding assumptions were the same as those for the SP-100 scalability study.

The four conversion systems were evaluated at power

levels of 2, 5, and 10 MWe, lifetimes of 3 and 10 years, and mission time frames of near term, mid term, and far term, incorporating technology enhancements with advancing time. Near-term technology consists of an 1,100 K peak temperature limit and an 8 kg/m^2 radiator specific mass. Mid-term technology corresponds to a 1,300 K peak temperature limit (1,800 K for thermionic) and a 6 kg/m^2 radiator. Far-term technology includes a 1,500 K peak temperature limit (2,200 K for thermionic) and a 5 kg/m^2 radiator. Mid-term technology was assumed to be appropriate for the reference time frame, 2010 to 2020.

Table 6.1.1-I provides specific mass results for these systems over the ranges of power level, lifetime, and mission time frame. The results indicate that mission time frame could have a significant effect on power system specific mass. From near-term to mid-term technology, as much as a 40 percent mass savings can be attained. The mass savings between mid-term to far-term technology ranges from 25 percent for Brayton systems to less than 5 percent for 10-year-life potassium Rankine systems.

Higher power level and shorter system lifetime have only a minimal effect on specific mass. The lower specific mass associated with shorter power system lifetime is mainly because of a decrease in the amount of reactor fuel and radiator armoring. Due to an economy of scale, power system specific mass also decreases with increasing power.

Of the candidate systems for the reference configuration (5 MWe, 10-year life, and mid-term technology), the in-core thermionic system was estimated to be the lowest mass option, followed by potassium Rankine, closed Brayton, and free-piston Stirling systems. However, the selection of a system must also take into account radiator area and relative system maturity. Although Brayton conversion has the largest radiator area, it is also the system that could be most readily developed for this application. This conclusion is based on the successful testing of Brayton units during the 1970s, selection of the Brayton system for solar dynamic power on Space Station Freedom, and the ease with which it can be scaled to higher power levels. The other three conversion sys-

TABLE 6.1.1-I.- NUCLEAR POWER SYSTEM SPECIFIC MASS RESULTS

Power, kWe	Life, years	Time frame	System specific mass (kg/kWe)			
			Brayton	Stirling	Potassium-Rankine	Thermionic
2,000	3	near term	26.4	25.5	17.8	-
		mid term	16.2	20.9	11.3	10.1
		far term	12.4	18.8	9.5	7.4
2,000	10	near term	29.3	28.1	18.9	-
		mid term	18.2	22.9	12.6	12.5
		far term	14.1	20.5	12.4	8.8
5,000	3	near term	24.1	24.2	17.2	-
		mid term	14.3	19.6	10.2	8.7
		far term	10.8	17.5	8.3	6.2
5,000	10	near term	26.8	26.6	18.1	-
		mid term	16.3	21.5	11.6	10.9
		far term	12.5	19.1	11.1	7.4
10,000	3	near term	22.6	22.6	-	-
		mid term	12.8	18.0	8.9	7.1
		far term	9.4	16.0	6.9	4.7
10,000	10	near term	25.2	25.0	-	-
		mid term	14.7	19.9	10.2	9.2
		far term	10.9	17.5	9.8	5.8

tems have major development issues associated with their suitability for a NEP application. The Brayton system was selected as the reference system because the radiator can be packaged on the 100 m boom within the established half angle, and it appears to be the most mature conversion option.

Mission Scenarios. Mission performance analyses were conducted to compare the power system options. The baseline mission scenario is a round-trip cargo delivery of 400 t from LEO (500 km circular) to Phobos orbit (6,000 km) and return of 100 t to GEO (35,800 km). A 2013 launch was assumed.

Several scenarios were investigated and compared with the baseline case. In addition to the 5 MWe reference power level, NEP-driven vehicles at 2 and 10 MWe were considered. Figure 6.1.1-1 shows transit time and mass to LEO for these options. The 2 MWe vehicle is the lowest mass option, but it would require a trip time nearly twice that for the 5 MWe case. The 10 MWe vehicle requires more mass to LEO (19 percent more than the reference case) but completes the mission in the shortest transit time (984 days). The 5 MWe reference vehicle appears to offer the best compromise.

6.1.1.2 Solar Electric Propulsion (SEP) Power System Options Assessment

Because of their advantages over both nuclear and chemical propulsion systems, SEP-driven vehicles could be a viable alternative. These advantages include the avoidance of nuclear reactor safety issues, flight experience, and reusability. In addition, all the technologies studied could be ready as early as 2000 to 2005.

Both photovoltaic (PV) and solar dynamic (SD) power systems were initially considered, but solar dynamic systems are too massive and were subsequently dropped. To provide some parity with a similar NEP conceptual design study, the power system was baselined to generate 5 MWe at Earth (1 au). The vehicle would be constructed in LEO, requiring the power system to be oversized such that, by passing through the Van Allen Belt, the power output would degrade to the rated 5 MWe level. The vehicle would spiral out from a 408 km altitude Earth orbit to a Mars transfer trajectory, rendezvous with Mars, spiral down to 17,033 km altitude (Mars-synchronous circular orbit), and deliver 400 t of payload to the surface. Because of the large degradation of power output due to radiation damage,

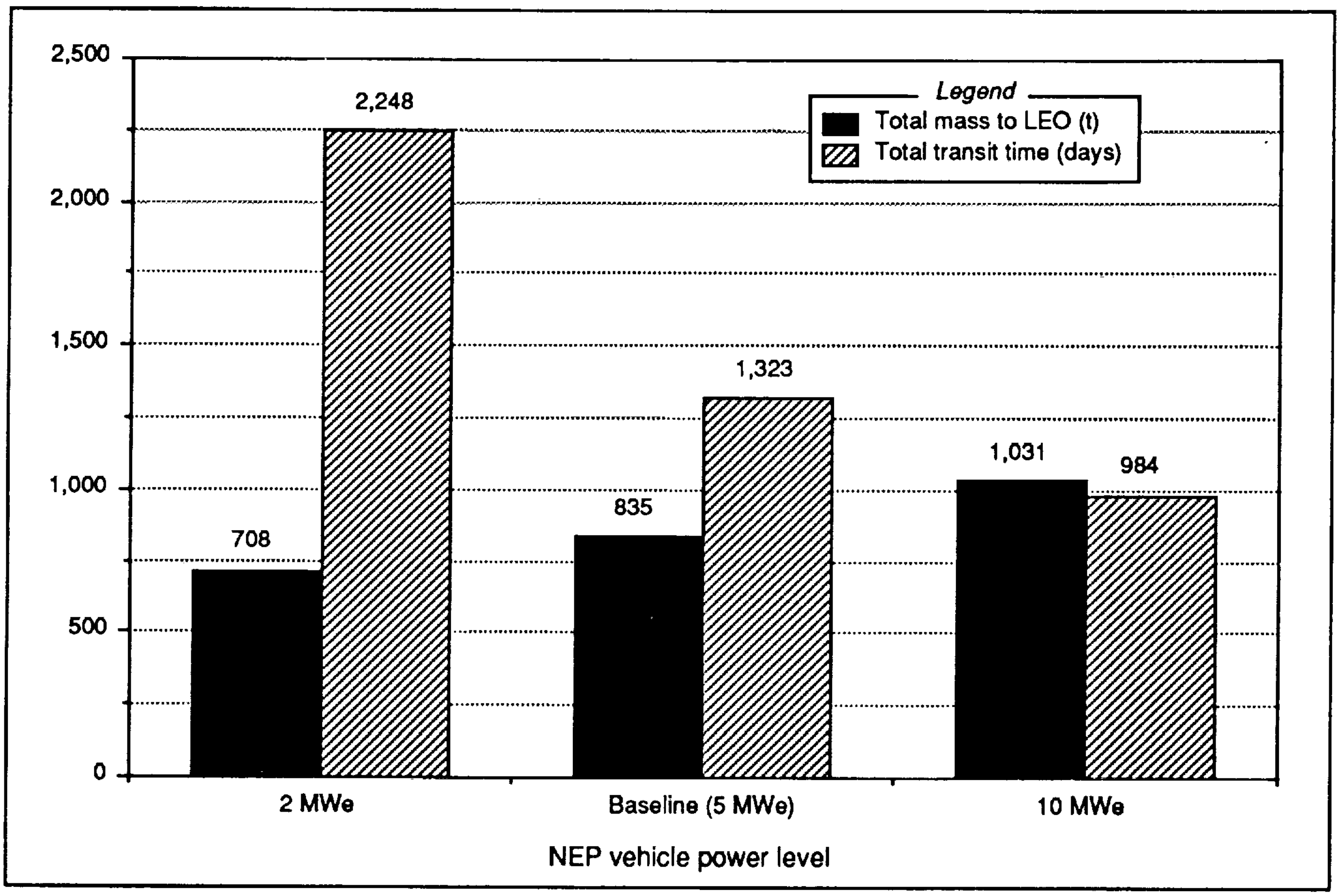


Figure 6.1.1-1.- Mass to LEO and transit time sensitivity to power level.

only one trip through the Van Allen Belt would be permitted for multiple trips to Mars. Vehicles returning to Earth would need to stay in high-Earth orbit, outside the radiation belts.

Power storage, which is required to thrust while in Earth's umbra, was not included in the power system. Storage is extremely massive and is not necessary for most of the trip to Mars. Without storage, however, the vehicle loses altitude as it passes through the shadow region. A trade study will be necessary to optimize the mass penalty of energy storage versus the propellant mass required for the extra trip time.

Propulsion system selection rationale followed that of the NEP study (see section 6.1.1.1). Thruster grouping changes were assumed to allow for variations in vehicle configuration between NEP and SEP vehicles.

Other considerations in defining a SEP system for comparison to other propulsion alternatives included selection of solar cells for mass, area, and radiation tolerance, and overall efficiency of the power generation and processing systems. Details of the findings of assessments in these areas can be found in Volume VI of this report.

The preliminary performance analyses performed during FY 1989 indicate mass savings over chemical and chemical/aerobrake systems for SEP vehicles comparable to the results for NEP. Specifically, a SEP vehicle could deliver 400 t to Phobos in 1,000 days requiring an initial mass of 650 t in LEO. Increasing this mass by 30 t would reduce the trip time to 870 days.

6.1.2 Surface Power

SP-100 Thermoelectric Lander for Lunar Surface Applications. During FY 1988, a concept for generating 825 kWe on the lunar surface using a SP-100 nuclear reactor was developed. Although it was an attractive concept for many reasons, assembly and construction of the plant presented a challenge. One way to obtain the same benefits (day/night availability, environmental compatibility, etc.) with less challenging deployment was sought. The SP-100 space nuclear reactor is being developed with packaging as a major consideration, and a concept for applying its advantages in this respect was developed. The objective of this study was to estimate the power system mass and total vehicle mass of an SP-100 nuclear reactor power system with thermoelectric converters integrated with a lunar lander.

This power system was designed to be the first significant power system (allowing for earlier low-power systems) at a lunar outpost. The SP-100 power system used here is a modified version of the current SP-100 baseline flight system. The required radiator area has been altered to account for higher heat sink temperatures. Using the same thermoelectric conversion used in the current development program, the power output is 100 kWe. The lander vehicle was assumed to be dedicated to the SP-100 power system.

After landing, radiators are automatically deployed from their stowed configuration, and power cables from the SP-100 thermoelectric power conversion system are manually routed down a landing strut terminating in a dc bus. A power system shunt load dissipator and an outpost interface module are manually positioned in a small excavation (on the order of 1 cu. ft.) in the lunar surface, which provides an in situ radiation shield. The main and secondary power buses are then manually deployed onto the lunar surface from the reactor to the outpost. A circumferentially shaped 4-pi radiation shield (i.e., one that shields in all directions except up and down) is integrated into the lander/reactor system and is man-rated at a distance of 1 km (i.e., 2.5 rem dose in 14 days) from the lander. Astronauts and scientists would, therefore, be able to visit the outpost during the 14-day lunar daylight period or for as long as a full lunar diurnal cycle, if the outpost is at least 1 km away. Longer stays would be possible if lunar regolith is used for radiation shielding or if the outpost is more than 1 km away. No maintenance is possible or required on the nuclear power system after system operation is initiated. The reactor power system and lander would remain at the original site at the end of the 7-year lifetime. The mass of the power system with shielding would range from 13 to 15 t.

The SP-100 thermoelectric lander could provide power sufficient for the early stages of evolutionary growth. The power that would be made available could shorten lunar outpost development time and supply immediate power for the human-tended phase. The lander could enable the immediate achievement of the first Lunar Evolution benchmark: the first occupation of a lunar outpost for a lunar night. The lunar outpost power system could be modularized to facilitate growth.

The period from landing the SP-100 system until full power is available could be quite short. Only a few hours are required to connect the power buses. A 24-hour start-up period is needed to thaw frozen coolant lines. A small portion of the outpost construction time would be required for the setup of the power system, enabling the crew to spend most of its surface stay constructing the outpost.

The SP-100 power system currently under development is being designed for use as a spacecraft power system; some changes would be necessary for surface applications. Significant modifications to the SP-100 reactor power system would be necessary for use on the Moon

because of the high daytime temperature and possible exposure to lunar dust.

Mars/Phobos/Deimos Power System Assessment. The martian system presents a unique set of challenges to power system design. An Earth-like diurnal cycle promises potential use of solar energy conversion on Mars, but dust storms and a more distant Sun work against that potential. The predominance of carbon dioxide in the martian atmosphere challenges the survivability of power system components at high temperatures. During FY 1989, the impacts of these characteristics on power systems design were assessed for sun-tracking PV power systems using regenerative fuel cells (RFC) for power storage.

For PV/RFC systems, Mars was found to provide as many opportunities as problems. Insolation varies with latitude; therefore, during the northern hemisphere summer, the Sun does not descend below the horizon at latitudes greater than 65 degrees north, and a power storage system might not be necessary during that season at those latitudes. Likewise, winter in the far southern latitudes offers almost no sunlight, making solar conversion systems impractical there. Martian dust suspended in the atmosphere also has mixed impacts; while it blocks direct radiation, it increases diffuse radiation. Therefore, selection of a sun-tracking versus a stationary solar array might be dependent on the expected frequency of dust storms for a given location.

6.1.2.1 Mobile Surface Power

Mobile power systems operate in the same environmental conditions as the stationary systems, but different design factors must be considered. For example, long-range rovers could require integrated or onboard power systems since they cannot be recharged from an outpost power supply. Construction or mining vehicles would probably remain near the site and could be powered by a dedicated system that would be periodically recharged by the outpost power system. The vehicle could remain for the required recharge period or exchange storage systems to increase the vehicle's utility value.

Integrated power systems are capable of long excursions or even global access for planetary exploration. Small scientific as well as larger pressurized crew rovers are candidates, and they may use photovoltaic or nuclear power. The nuclear options include radioisotope thermoelectric generators (RTGs), dynamic isotope power systems (DIPS), and reactors. The systems can be compared by quantifying various figures of merit for each technology over a range of power levels and operating times.

The matrix of power system domains shown in figure 6.1.2-1 is quantified by power level and operating time. Overlaid on the matrix is a set of rover applications defined by surface operations scenarios and other exploration scenarios. The power system technologies are shown within the ovals depicting each shaded area, and the rover applications are in the rectangles. The boundaries of the shaded areas are regions where the systems have equivalent mass estimates. These boundaries would be slightly adjusted based on planetary location and the particular power system component technology selected. Table 6.1.2-I describes the matrix in further detail and gives the rationale behind the selected technology areas.

Power system studies were performed (some continuing) for three rover applications: (1) a 30 kWe Mars crew exploration vehicle, (2) a 0.5 kWe robotic explorer, and (3) a 16 kWe Mars crew sortie vehicle. These studies are discussed below.

Nuclear-Powered Mars Rover. Human exploration of Mars could expand beyond the areas adjacent to the landing site with a combination habitat/mobile laboratory with sufficient power to support the crew, science activities, and mobility. A typical set of vehicles is shown in figure 6.1.2-2. The habitation cab houses the auxiliary power supply for emergency return to the landing site (return ascent vehicle) in the event of a main power system malfunction. One other rescue scenario would be to leave the ascent vehicle in orbit and land it near the rover. The auxiliary power system consists of photovoltaic deployable arrays and RFCs. The mass of the arrays is a strong function of the collection time. The minimum array area and mass are obtained when recharging occurs during maximum allowable daylight and return travel is limited to nighttime only. The mass of the RFCs is mainly devoted to life support needs and is relatively insensitive to mobility power required for the base case of 6 hours travel per night. The rover mass is a function of speed, particularly as the mass of the power cab increases. The total rover mass, including the reactor power cab with shield, is approximately 26,000 kg at a cruising speed of almost 20 km/hr.

Isotope Rover Power System. Smaller rovers will be used to perform science and robotic missions. The projected power levels for these rovers are in the 0.5 kWe to 1 kWe range. RTG and DIPS power systems were compared at the 0.5 kWe level. Figure 6.1.2-3 compares power system mass for the General Purpose Heat Source and modified RTG and for two levels each of the closed Brayton cycle and the free piston Stirling engine. Both DIPS systems with redundant power conversion units are comparable on a mass basis with the advanced modified RTG. Plutonium (a space-qualified isotope) is and will continue to be in limited supply. High conver-

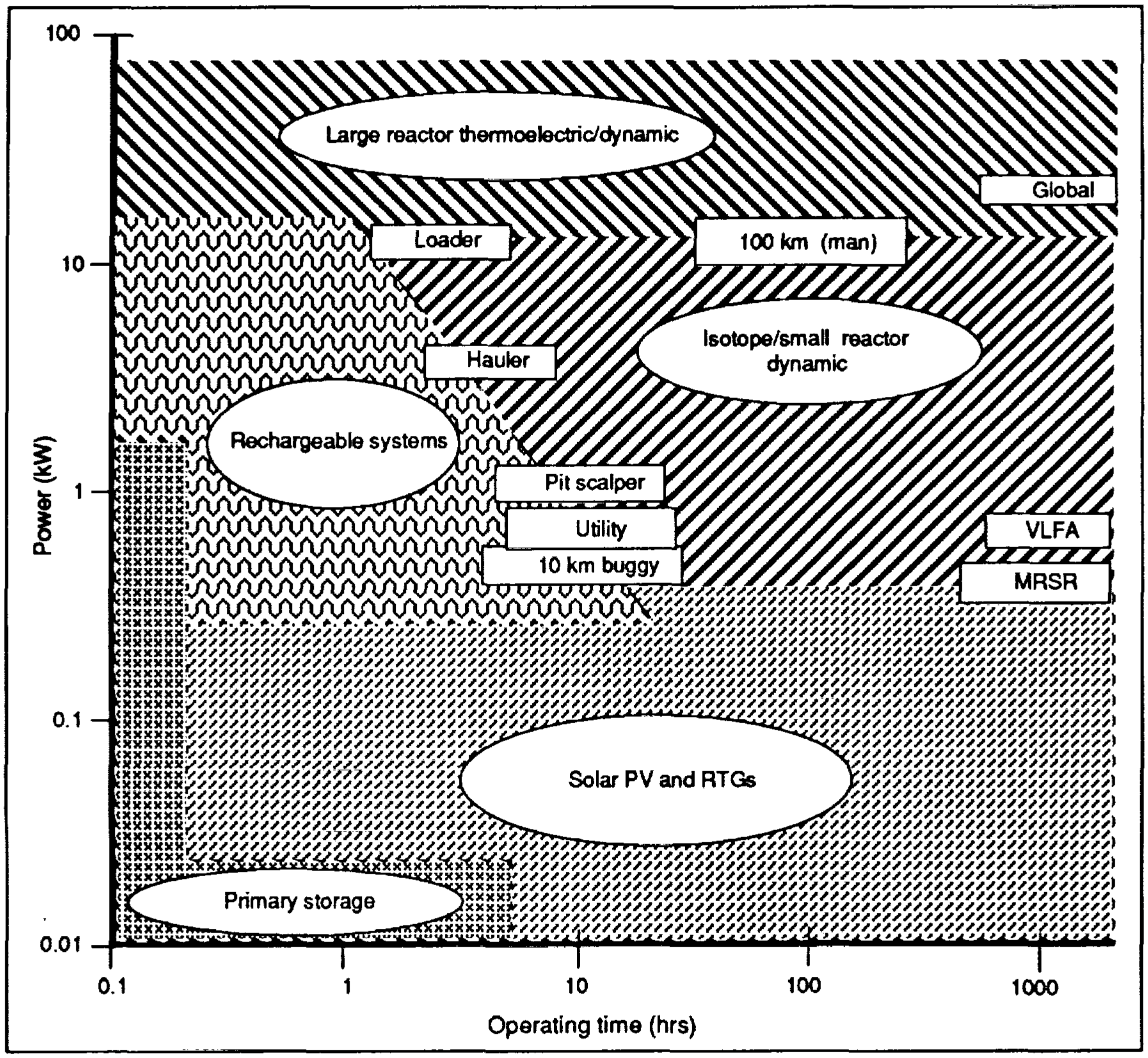


Figure 6.1.2-1.- Rover power system domain matrix.

sion efficiency of the dynamic systems offers significant reduction in the amount of isotope required to produce power as shown in figure 6.1.2-4.

The dynamic systems scale favorably to the higher power levels as shown in table 6.1.2-II, which provides data for a system design for a 16 kWe crew rover with a 100 km range. The isotope savings are considerable, and the mass is reduced to about 20 percent of the RTG power system. Power system volume is also significantly reduced. Thus, technology development in dynamic systems can be used over a wide range of power levels applicable to a variety of rover applications.

Solar Photovoltaic Mars Rover. Another option for supplying rover power is PV cells. Even though the dynamic systems reduce the strain on our plutonium inventory, many robotic rovers deployed to characterize planetary surfaces could add up to a major burden on isotope production. If rovers use body-mounted arrays, the configuration must account for the array location and reduce shadowing from ancillary equipment. The use of deployable/retractable arrays would probably require the rover to be stationary during charging of the energy storage device.

Results indicate that optimum performance of a solar-powered rover is between ±30 degrees latitude. High-efficiency arrays (> 30 percent) would greatly enhance the rover power system performance by reducing array area; this is not as important for stationary systems, where area is a secondary factor.

PV power could be a viable option for a Mars rover even with dust storm occultations, since scattered light can make up a significant proportion of the array output. The worst case point design considered 20 We/m^2, and advanced solar cells would yield almost 30 We averaged over the martian day. During dust storms, the array output is reduced by only 30 percent. Future efforts will quantify the mission profile and the effect of different energy storage devices on the power system configuration as well as vehicle integration issues.

TABLE 6.1.2-I.- ROVER POWER DOMAIN DESCRIPTION

Power system	Comments	Applications
Primary storage	One-time use; (bounded by mass) (time * power * Whr/kg)	Apollo-type missions
PV only	Body-mounted array; no night-time operations (bounded by power); limited use	Robotic exploration and mining
PV and storage	Body mounted and/or deployable/ retractable array; night plus peak power storage bounded by mass (time * power * Whr/kg) and array area	Robotic exploration
Rechargeable storage	Multiple use; power level and time limited by mass of system	Local pressurized transport; utility; construction mining
Isotope/reactor dynamic	Continuous use, unlimited range (bounded by isotope inventory, power); some mass constraints	Robotic rover for global range
RTG	Continuous use; unlimited range (bounded by isotope inventory, power)	Robotic rover
Reactor	Continuous use; unlimited range (bounded by reasonable mass)	Heavy mining; pressurized transport (manned, global range)

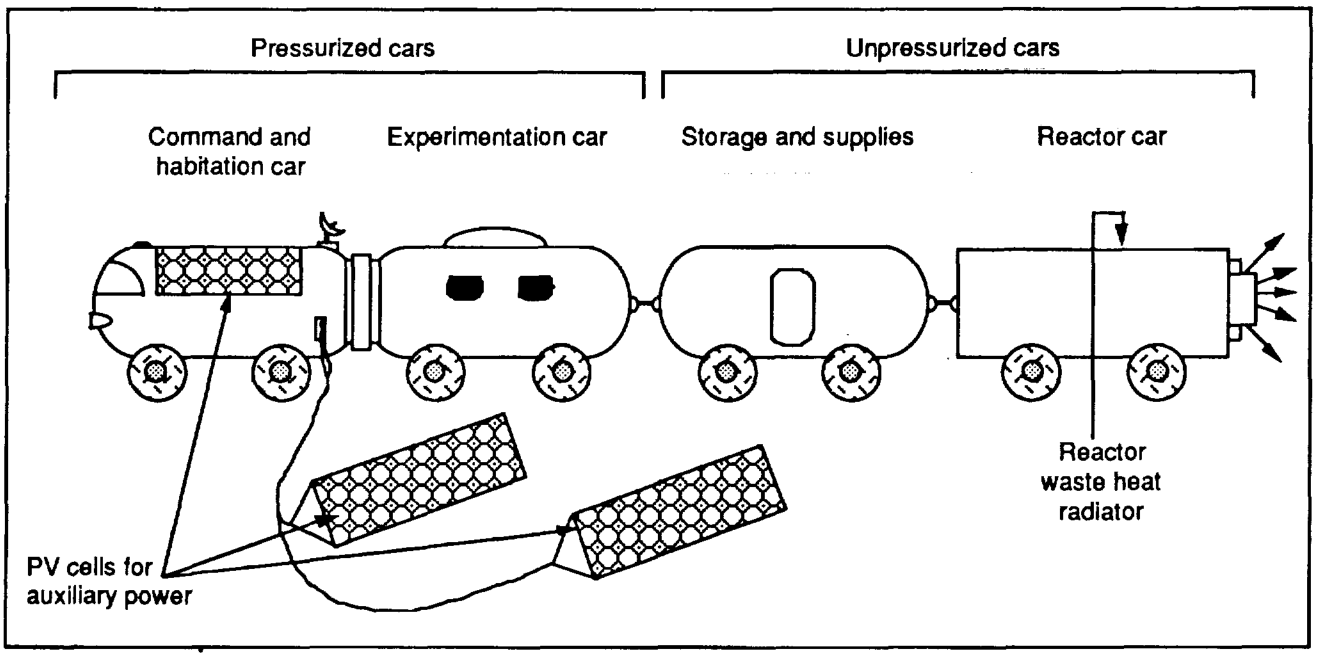


Figure 6.1.2-2.- High power rover concept.

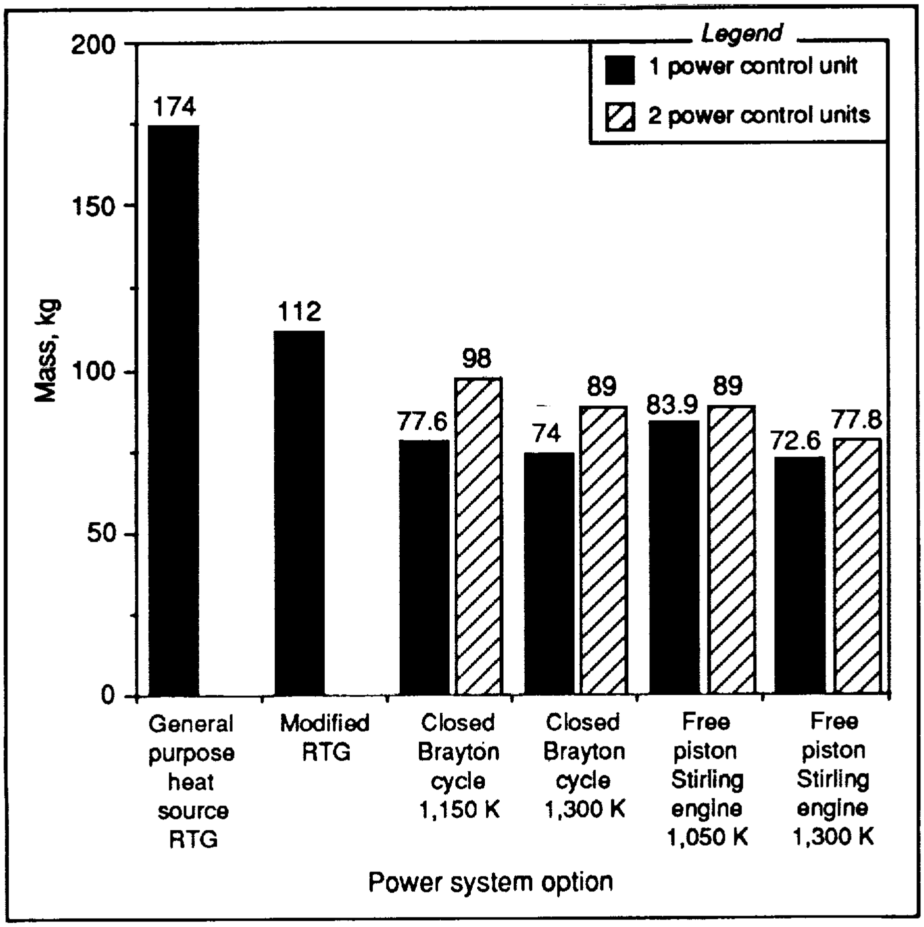


Figure 6.1.2-3.- Isotope power system mass comparison for 500 We.

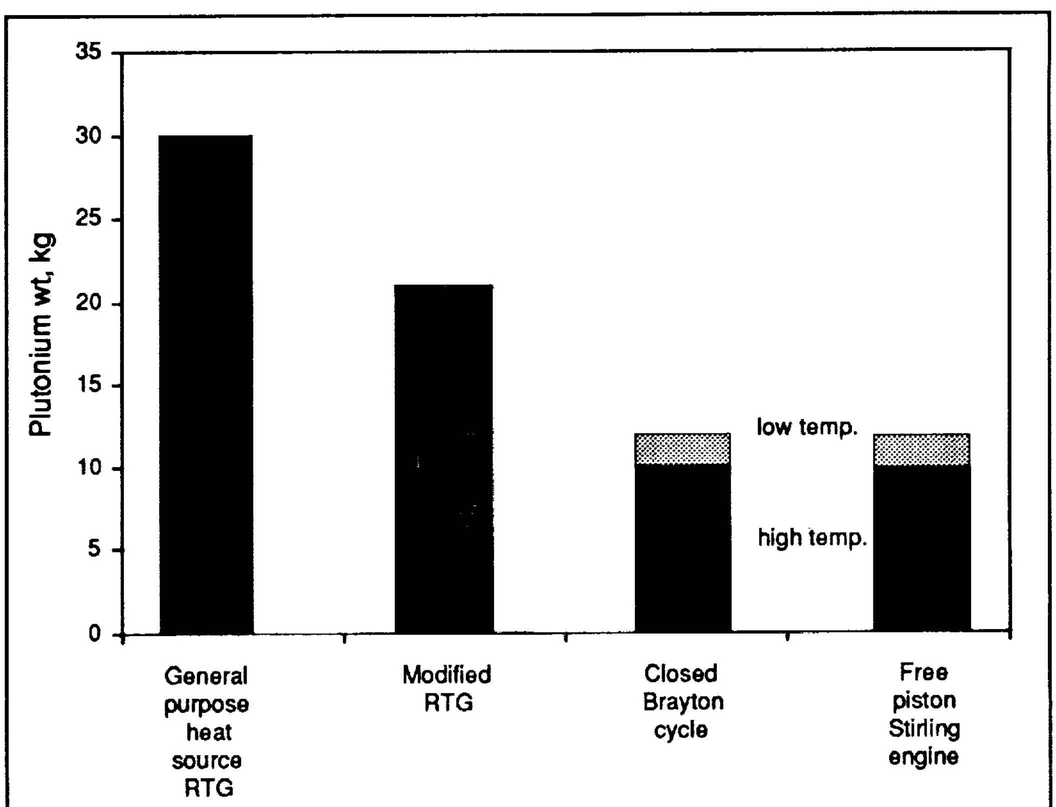


Figure 6.1.2-4.- Plutonium inventory for 500 We isotope systems.

TABLE 6.1.2-II.- ISOTOPE POWER SYSTEM ANALYSIS FOR MANNED MARS ROVER (16KWE)

	RTG	Brayton
GPHS blocks	1,570	278
Heat in	362 kWt	64 kWt
Radiator area	70 m^2	19 m^2
Volume	10 m^3	1.0 m^3
Mass (kg)		
GPHS blocks	2,280	304
Radiation and housing	1,700	107
Structure (15%)	600	60
Engines	–	283
Power conditioning	165	165
Total	4,745	919

6.1.3 Human Nuclear Radiation Issues

The issues and ramifications of human proximity to nuclear systems in the context of space exploration and its natural radiation hazards were identified across the entire spectrum of human presence across a complete range of manmade nuclear sources. Humans can safely function in all instances of proximity to space nuclear systems investigated to date, provided that sensible safety measures are taken. Further, it seems these safety measures can be achieved reasonably. No conclusions have been reached about the possibility of direct human servicing of recently operated reactors. Natural radiation hazards appear more menacing than manmade hazards, yet both demand the utmost respect. Rather than being part of the radiation problem, nuclear systems may be an important part of systems-oriented solutions to overall human radiation issues.

Two major considerations in planning human presence in the vicinity of nuclear power systems are the characteristics of radiation from nuclear reactors and the standards for human exposure to such radiation. Reactor-produced radiation characteristics are comparable to natural space environment radiation; similar types of radiation (neutron and gamma) are encountered. However, higher energy radiation and more penetrating radiation (charged nuclei) are present in the natural space environment. Standards for human exposure to radiation are getting tighter; the National Council on Radiation Protection has proposed new limits (expected to be adopted by NASA) of 100-400 rem career dose limit and 50 rem annual and 25 rem monthly limits. Tight standards for human exposure will mandate significant protection measures against the natural space radiation environment. Therefore, reactor-produced radiation is not expected to be a significant consideration in most human exploration mission designs.

Table 6.1.3-I summarizes the assessments to date of the issues involved in planning use of nuclear reactor power systems for human exploration. All situations assessed to date are safe, or can be made safe through appropriate reactor system and operation design. These assessments have revealed two important conclusions. First, nuclear reactor power systems, because they offer greater mass leverage, and in some cases can reduce exposure time (through faster transportation), may be viewed as a solution to human sensitivity to radiation. Mass leverage means that more mass can be added for shielding against the natural space environment at a lower cost. Reduced exposure time can mean more flexibility and utility of human crews in exploration planning. Second, shielding for the natural space radiation environment can be synergistic with that for nuclear reactor power systems. Human exploration systems designers should recognize and exploit the potential mass savings offered through dual use of shielding mass.

6.2 PROPULSION SYSTEMS SPECIAL ASSESSMENT

Studies conducted during FY 1989 by the Propulsion Systems Special Assessment Agent focused on comparing propulsion systems and quantifying the advantages of their use for transportation to Mars.

The Advanced Propulsion Comparison Study compared chemical rockets with nuclear thermal rockets (NTRs), both with and without aerobrakes, for the Mars Expedition and Evolution case studies. Both chemical/aerobrake and NTR systems show a large benefit (in some cases more than 50 percent) over all-chemical propulsion. Chemical/aerobrake and NTR systems are competitive, with NTR showing advantages for the Mars Expedition case, and chemical/aerobrake showing advantages for the Mars Evolution case under the conditions considered. Variables such as aerobrake efficiency (mass fraction) and trajectory/mission design strongly affect the leverage of propulsion systems. NTR/aerobrake has the highest leverage of systems included in this study for all missions considered.

The Future Propulsion Technology Study examined technologies judged to be beyond advanced chemical, aerobrake, NTR, NEP, and SEP, but available within the 25- to 30-year time frame of interest for use in Mars cargo missions. These included solar sails, very high-power NEP and SEP, solar and laser thermal propulsion, rail guns, mass drivers, and tethers. Of these, solar sails and very high power NEP and SEP warrant further study for broad application to exploration missions.

TABLE 6.1.3-I.- HUMAN NUCLEAR RADIATION ISSUES

Increasing human presence →

Radiation source	Service at power source	Brief service (hrs.) in proximity	Extended activity (wks.) in vicinity	Habitation in vicinity	Habitation on board
Radioisotopes	O	O	O	O	O
Small reactors < 1 MW t	SD	SD	O/SD	O	O
Large reactors ≥ 1 MW_t	SD	Node: SD	Node: O/SD	Node: O	Node: O
		Surface: SD	Surface: O/SD	Surface: O	Surface: N/A
Nuclear electric vehicle	SD	SD	SD	O/SD	O
Nuclear thermal vehicle	SD	SD	SD	O/SD	O

Safe under standard practices.

Safe if and when special precautions are taken.

Hazards to be avoided.

O = Operating
SD = Shut down

No pattern: No conclusion is stated.

Special concerns.

A third study, the Fuel Systems Architecture Study, was conducted in coordination with the Office of Space Station, the Office of Space Flight, and the Office of Aeronautics and Space Technology, to identify fuel systems architectures to support transportation vehicles for exploration missions and other requirements. Preliminary results obtained in FY 1989 are expected to have broad implications for other options, and will help reduce the number of architecture options that need to be investigated.

6.2.1 Nuclear Thermal Rocket/Chemical Aerobrake Comparison for Mars Transfer Propulsion

The objective of this study was to quantify and compare the performance capability of the NTR and chemical propulsion systems, with and without aerobraking, for a selected set of Mars mission opportunities in the 2000 to 2020 time frame. Studies conducted in FY 1988 and earlier by Lewis Research Center's (LeRC) Advanced Space Analysis Office (ASAO), Marshall Space Flight Center (MSFC), and others indicated that aerobraked chemical and all-propulsive NTR systems each had the potential to reduce initial mass in low-Earth orbit (IMLEO) for split cargo/piloted sprint missions to Mars by approximately 50 percent or more compared to an all-propulsive chemical mission. Because of the different ground rules and assumptions made in the various investigations, drawing overall conclusions regarding relative system performance is difficult. To help clarify the performance issue, a consistent performance comparison study was conducted of NTR and chemical propulsion systems. Since results were to be compared with other studies being performed for OEXP, study ground rules were coordinated with other NASA field centers and contractor organizations supporting OEXP; these are discussed in detail in Volume VI of this report. The principal difference between this study and the MSFC study is the inclusion of the effect of gravity losses. Also, the MSFC/MMC baseline chemical/aerobrake system assumed multiple, expendable Shuttle-Z type trans-Mars injection (TMI) stages, whereas this analysis assumed a single TMI stage with either single or multiple engines/propellant modules to achieve near-optimal thrust to initial weight ratios for each of the propulsion system options under consideration.

Two mission scenarios were selected and investigated. In the first, Mars Evolution, opposition- and conjunction-class trajectories are used to establish and support a permanently inhabited outpost. Short-duration, high-energy trajectories were also studied to reduce the duration of the interplanetary travel time and thus increase mission time at Mars. In the second, the Mars Expedition strategy is used: an unmanned cargo vehicle (carrying all hardware to be delivered to Mars, and in some cases, the fuel for the piloted vehicle's Mars-to-Earth return trip) is launched on a slow, one-way, relatively low-energy conjunction-class trajectory to Mars. To limit the time the crew is exposed to the space environment, the piloted vehicle uses a higher energy, opposition-class "sprint" trajectory.

Three NTR performance levels were investigated. The first, designated "1972 NTR," represents 1972-vintage NERVA (75,000 lbf thrust) and Phoebus-2A (250,000 lbf thrust) nuclear rocket performance capability with graphite matrix/composite reactor fuel elements providing an engine specific impulse of 900 seconds. The second performance level, referred to as "1989 NTR," represents the performance of similar size engines built with state-of-the-art materials and propulsion system components that would provide an increased engine thrust-to-weight ratio. The third performance level, referred to as "advanced NTR," assumes that the 1989 NTR materials and propulsion systems, along with advanced carbide fuel elements, will provide even higher thrust-to-weight ratio and increased specific impulse. A propulsion system using both the 1972 performance level engines and aerobraking was also investigated; it is referred to as NTR/AB. The key propulsion system and aerobrake assumptions used in this study are summarized in table 6.2.1-I.

The principal figure of merit for comparing the propulsion systems is the required IMLEO. This figure of merit has been used frequently in previous studies, and it continues to be of interest because of its association with program costs and mission complexity. To determine the IMLEO, ASAO's vehicle sizing code was utilized.

6.2.1.1 Mars Evolution Case Study

To provide a representative sampling of the seven flights developed for the case study, the first (opposition-class

TABLE 6.2.1-I.- PRINCIPAL PROPULSION SYSTEM AND AEROBRAKE SIZING ASSUMPTIONS

a) Engine systems

Engine Systems:	Propellant	Isp, seconds	Thrust, klbf	Engine mass, t	Shield mass, t
SSME derivative	LOX/LH_2	480	532	3.6	N/A
RL-10 derivative	LOX/LH_2	471	20	0.2	N/A
1972/1989 NTR	LH_2	900	75	11.3/5.5	4.5
1972/1989 NTR	LH_2	900	250	19.0/15.8	9.0
Advanced NTR	LH_2	1,000	75	5.5	4.5
Advanced NTR	LH_2	1,000	250	15.0	9.0
Auxiliary Chemical	Storable bipropellant	310-316	Low	Low	N/A

b) Aerobrake mass for various vehicles
[Calculated as a percentage of braked payload mass]

Vehicle	Relative L/D ratio	Aerobrake mass, %
Evolutionary (piloted)	Low	13.2
Expeditionary cargo	Medium	17.5
Expeditionary piloted	Medium	20.0

in 2004), fifth (conjunction-class in 2011), and seventh (quick mission in 2016) flights were selected for detailed study.

IMLEO requirements for chemical and NTR propulsion systems operating both all propulsively (AP) and with aerobrakes (AB) were determined for the 2004 and 2011 missions; several trades were also performed. The effect of ΔV optimization on the IMLEO for the AP configurations was investigated for the 2004 mission. The aerobrake mass fractions required to provide chemical/AB IMLEOs comparable to those of all-propulsive NTR vehicles were determined for both the 2004 and 2011 missions. The mass penalty associated with recovery of the Earth-departure stage was also addressed. Finally, the IMLEOs required to support quick (one-way trips of less than 6 months duration) piloted missions to and from Mars were determined.

The results for the 2004 and 2011 reference missions (table 6.2.1-II) show IMLEO for the baseline chemical/AB system to be 573 to 662 t respectively. The increased mass for the 2011 mission is attributed primarily to the additional propellant needed to achieve the 6,000-km circular Phobos parking orbit. For the all-propulsive systems, recovery of the core spacecraft results in a major mass penalty. Compared to the baseline chemical/AB system, chemical/AP is between 6.6 and 4.8 times more massive, even with discrete stages for each of the main propulsive burns. This comparison illustrates quite dramatically the benefit that may be realized by chemical systems if a single, common aerobrake can be developed for use at both Mars and Earth.

The baseline Mars Evolution scenario requires IMLEO values for the 1972 NTR that are between 2.0 and 1.4 times heavier than the chemical/AB system. For the 1989 NTR system, these numbers decrease to the range between 1.8 and 1.3. Greater mass savings are realized with the advanced NTR system; the IMLEOs are approximately 1.4 times heavier than the baseline chemical/AB system for the 2004 mission and essentially the same for the 2011 mission. Finally, the NTR/AB system has the lowest IMLEO of all the systems studied, with masses ranging from 380 to 443 t.

Because the SRD-specified ΔV budgets were optimized for aerobraked systems (with large ΔV increments for orbit capture maneuvers), a similar ΔV budget optimized for all-propulsive systems was also determined. By minimizing the total ΔV for the 2004 mission, substantial mass reductions are possible for the all-propulsive systems, ranging from 690 t for the chemical/AP system, to 145 t for the 1972 NTR, to 82 t for the advanced NTR.

TABLE 6.2.1-II.- MARS EVOLUTION CASE STUDY – MISSION AND PROPULSION SYSTEM COMPARISON

Case	Stage propulsion: Trans-Mars injection	Stage propulsion: Mars orbit capture	Stage propulsion: Trans-Earth injection	Stage propulsion: Earth orbit capture	IMLEO (t) 2004	IMLEO (t) 2011	Percent of chemical/AB IMLEO 2004	Percent of chemical/AB IMLEO 2011
Chemical/AB	C_1	AB_1	C_2	AB_1	573	662	100%	100%
Chemical/AP	C_1	C_2	C_3	C_4	3,800	3,141	663%	475%
1972 NTR	N_1	N_2*	N_2*	N_2	1,133	933	198%	141%
1989 NTR	N_1	N_2*	N_2*	N_2	1,031	857	180%	129%
Advanced NTR	N_1	N_2*	N_2*	N_2	787	680	137%	103%
NTR/AB	N_1	AB_1	N_2	AB_1	380	443	66%	67%

*Tanks staged
Note: Subscripts indicate discrete stages.

The sensitivity of IMLEO to aerobrake mass was examined for the 2004 and 2011 flights. By varying the aerobrake mass fraction, the cross-over points of comparable IMLEO between the aerobraked and all-propulsive NTR systems were determined. Results for the 2011 mission (figure 6.2.1-1) show that aerobrake mass, as a percentage of payload, can range from approximately 13 to 26 percent and still maintain advantages over other systems. These data also show the mass reductions possible in going from 1972 NTR to advanced NTR performance levels.

The mass penalty associated with recovering the TMIS in addition to the basic core spacecraft was also studied for the 2004 and 2011 flights. The basic recovery profile assumes a retro burn by the TMIS after insertion of the MTV on its interplanetary trajectory. This retro burn places the TMIS on a 24-hour elliptical orbit with a final circularization burn returning the stage to its original 500 km parking orbit about Earth.

For the chemical systems, recovery of the TMIS results in a mass penalty ranging from 22 to 37 percent, whereas the range for the NTR systems varies from 24 to 32 percent for the 2004 mission and from 17 to 23 percent for the 2011 mission (see Volume VI of this report for more details). A second recovery option examined for the all-propulsive NTR systems involved using a single Phoebus-class engine with staged tanks to perform the entire round-trip mission. The corresponding mass penalty associated with this recovery option is significantly reduced, and it ranges from 7 to 10 percent for the 2004 mission, and from 9 to 11 percent for the 2011 mission.

The seventh flight in the evolutionary case study departs Earth in 2016 and is intended to initiate the operational phase of the Mars outpost by extending the number of days available to the crew for Mars surface operations. For a given mission duration, this is accomplished by reducing the interplanetary transit times. Figure 6.2.1-2 compares advanced propulsion systems and their IMLEO requirements as a function of one-way trip time. Propellant for the Earth return trip is assumed to be provided at Phobos either via in situ propellant produc-

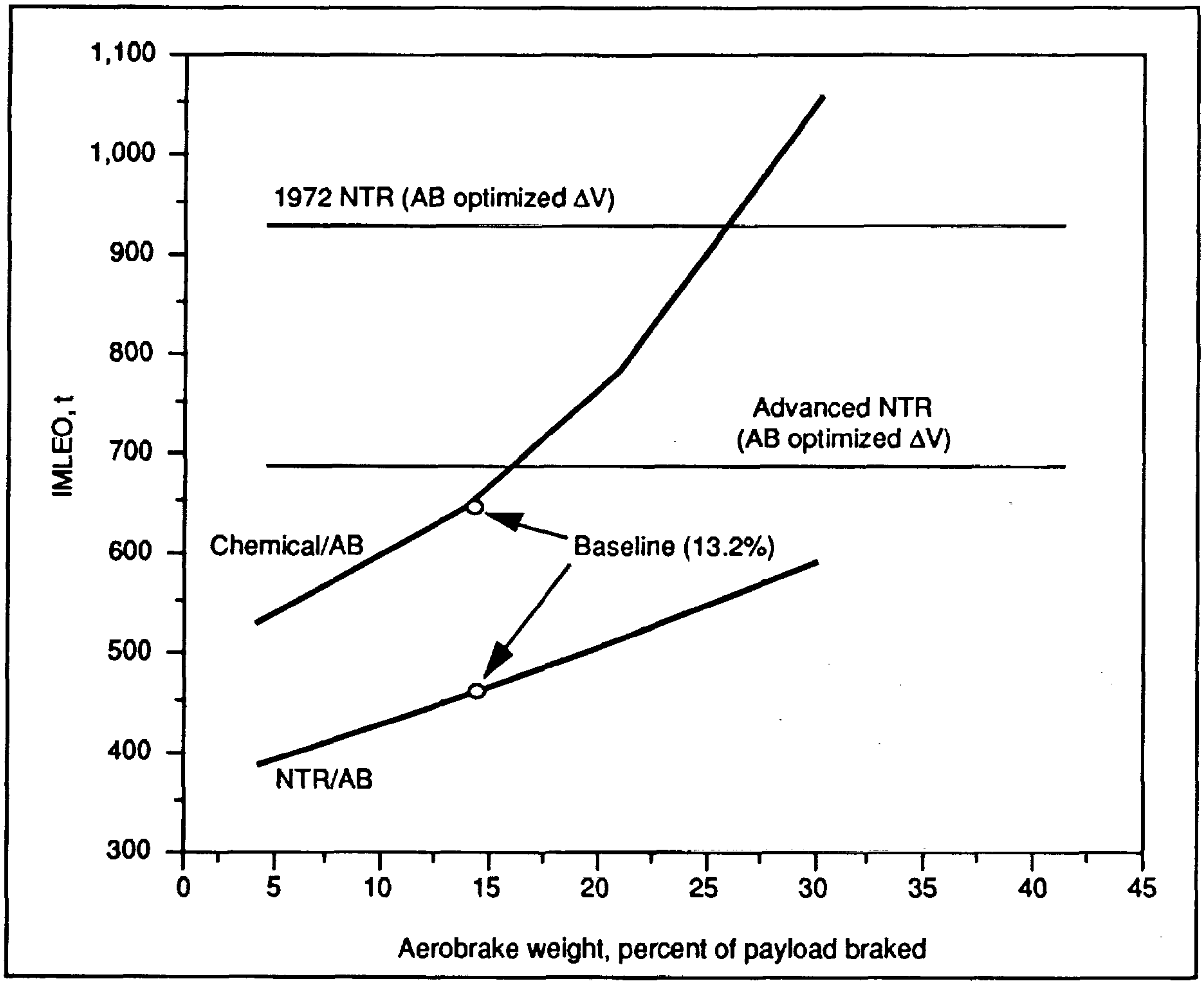


Figure 6.2.1-1.- Crossover IMLEO values for aerobraked and "all propulsive" NTR systems (2011 conjunction mission – fifth flight).

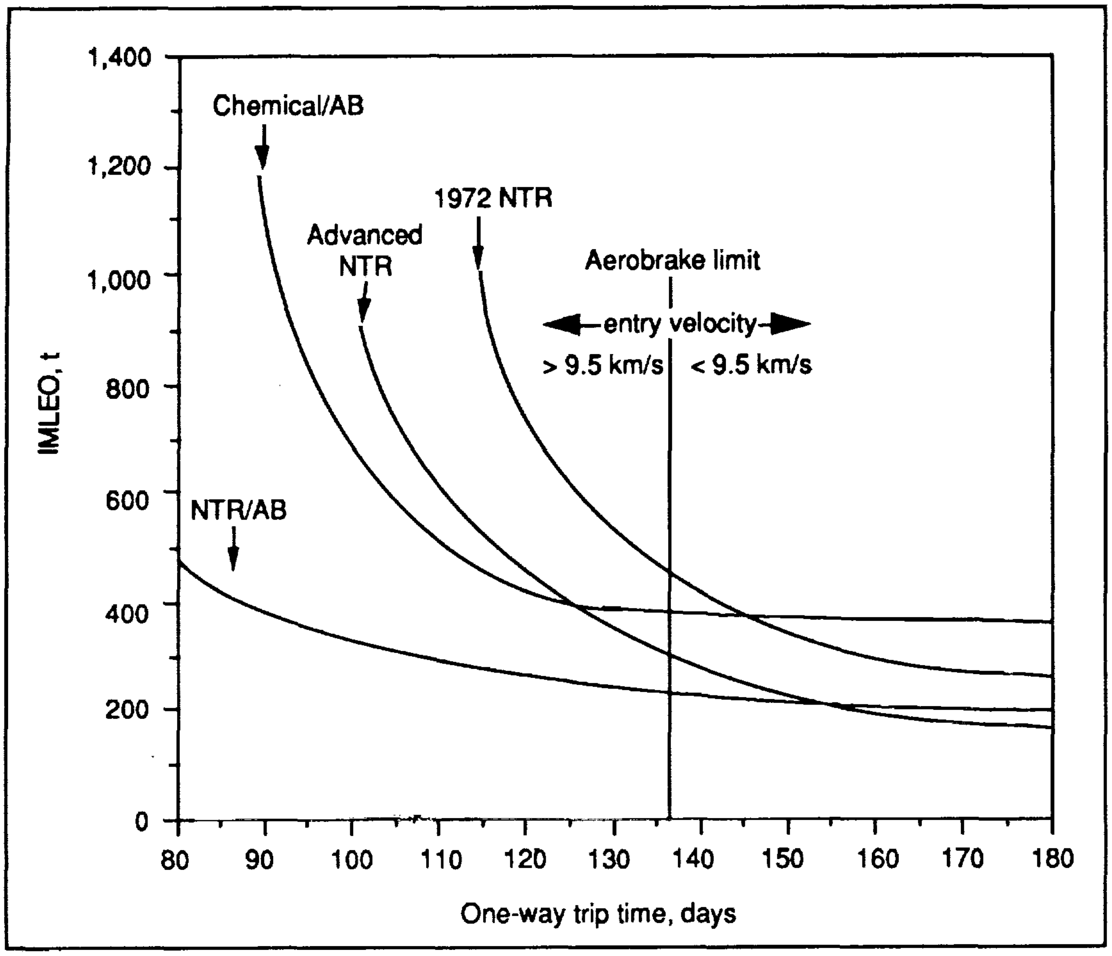


Figure 6.2.1-2.- Earth to Mars "quick trip" propulsion comparison for the 2016 flight (without staged tanks, Phobos orbit, 5 crew).

tion at the gateway or fuel transfer from NTR/NEP cargo vehicles operating in the "Hohmann tanker mode."

Both the Earth-to-Mars and Mars-to-Earth legs were analyzed for the same transit times to ensure that the TMI/Mars orbit capture tankage for the MPV can accommodate the return propellant requirements. One-way trips as short as 80 days and as long as 6 months were examined. As indicated, quick missions to Mars of about 120 days appear to be possible with all-propulsive NTR systems for IMLEO values ranging from about 490 t (advanced NTR) to 750 t (1972 NTR). Shorter trip times are also indicated, but with the steep rise in mass, propellant loadings are expected to become prohibitive. By extending the one-way trip times from 4 to 6 months, the IMLEO for the all-propulsive NTR systems is reduced to less than 300 t. The basic vehicle configuration would consist of a manned mission module, a propellant tank of about 140 t LH_2 capacity, and a single Phoebus-class NTR.

Aerobraked chemical and NTR systems appear to be capable of very short trip times (80 days for an NTR/AB system with IMLEO less than 500 t) if it were not for the large g-loads and entry velocities encountered at Mars. The specified entry velocity limit of 9.5 km/s restricts one-way trip times for aerobraked systems to about 140 days, although a combination of propulsive and aerodynamic braking may make shorter trip times possible.

6.2.1.2 Mars Expedition Case Study

Trade studies performed with respect to the Mars Expedition case study include (1) quantify the IMLEO requirements for chemical and NTR propulsion systems operating all propulsively and with aerobrakes, (2) determine the effect of the Mars parking orbit, (3) examine launch opportunity sensitivity for four split/sprint mission opportunities in the 2000-2010 time frame, and (4) assess the sensitivities associated with variations in the mission mode. Since detailed and comprehensive discussion of the results of these trades is not possible here (see Volume VI of this annual report), only highlights are covered.

The 2002 mission was used in the propulsion system

comparison trade (see table 6.2.1-III). The baseline chemical/AB system has a combined total mass for both the piloted and cargo vehicles of 686 t (with g-losses included). The all-propulsive (AP) chemical system is 2.5 times more massive, with an IMLEO of 1,782 t. The all-propulsive 1972 NTR has an IMLEO of 675 t, which is approximately 98 percent of the baseline chemical/AB system. The 1989 NTR and advanced NTR systems have IMLEOs of 87 and 72 percent of the baseline chemical/AB respectively. The NTR/AB system is the lightest of all the propulsion options examined, with an IMLEO about 40 percent less than the chemical/AB baseline.

The sensitivity of IMLEO to variations in the Mars parking orbit is discussed in detail in Volume VI of the Exploration Studies Technical Report. Briefly, the results indicate that orbit selection does not strongly affect the comparisons between aerobrake and all-propulsive systems, although it does affect the IMLEO requirements.

To determine the ability of the various propulsion options to capture other mission opportunities in the 2000 to 2010 time frame without major mass penalties, the IMLEO requirements for the available opportunities were calculated and compared. The 2004 and 2007 mission options have substantial Mars orbit capture ΔV requirements (about 5.6 km/s for 2004 versus about 4.6 km/s for 2002), which lead to substantial mass increases for the all-propulsive systems, especially the chemical option. Results show that the 1972 NTR system has a slightly lower IMLEO than the chemical/AB system for the 2002 and 2010 opportunities, with this trend reversing for the 2004 and 2007 opportunities. The advanced NTR has a lower IMLEO than either the chemical/AB or the 1972/1989 NTR options; the NTR/AB option shows the lowest IMLEO of all the systems considered.

Due to concern over the ability to transfer propellant in Mars orbit from the Mars cargo vehicle (MCV) to the Mars piloted vehicle (MPV), the FY 1989 baseline split mission assumes that the trans-Earth injection (TEI) stage is flown on the MPV, leaving a significantly smaller amount of payload (primarily the Mars descent/ascent vehicle) to be carried by the MCV. Three mission mode options — the FY 1989 baseline split, the traditional split, and an all-up mission — are compared in figure 6.2.1-3. The baseline split and all-up mission modes show comparable IMLEO values for the advanced concepts, indicating an advantage to the all-up mission mode from the standpoint of mission simplification (one versus two vehicles). The traditional split shows the lowest IMLEO values for the mission modes considered. In terms of

TABLE 6.2.1-III.- MARS EXPEDITION CASE STUDY – MISSION AND PROPULSION SYSTEM COMPARISON

Case	Stage propulsion: Trans-Mars injection	Stage propulsion: Mars orbit capture	Stage propulsion: Trans-Earth injectiion	IMLEO (t): Piloted	IMLEO (t): Cargo	Total IMLEO	Percent of chemical/AB IMLEO
Chemical/AB	C_1	AB	C_2	547	–	686	100%
	C_1	AB	–	–	139		
Chemical/AP	C_1	C_2*	C_2	1,494	–	1,782	260%
	C_1	C_2	–	–	288		
1972 NTR	N_1	N_2*	N_2	499	–	675	98%
	N_1*	N_1	–	–	176		
1989 NTR	N_1	N_2*	N_2	439	–	597	87%
	N_1*	N_1	–	–	158		
Advanced NTR	N_1	N_2*	N_2	356	–	496	72%
	N_1*	N_1		–	140		
"Clean" NTR/AB	N_1	AB	N_2	308	–	421	61%
	N_1	AB	–	–	113		

*Tanks staged
Note: Subscripts indicate discrete stages

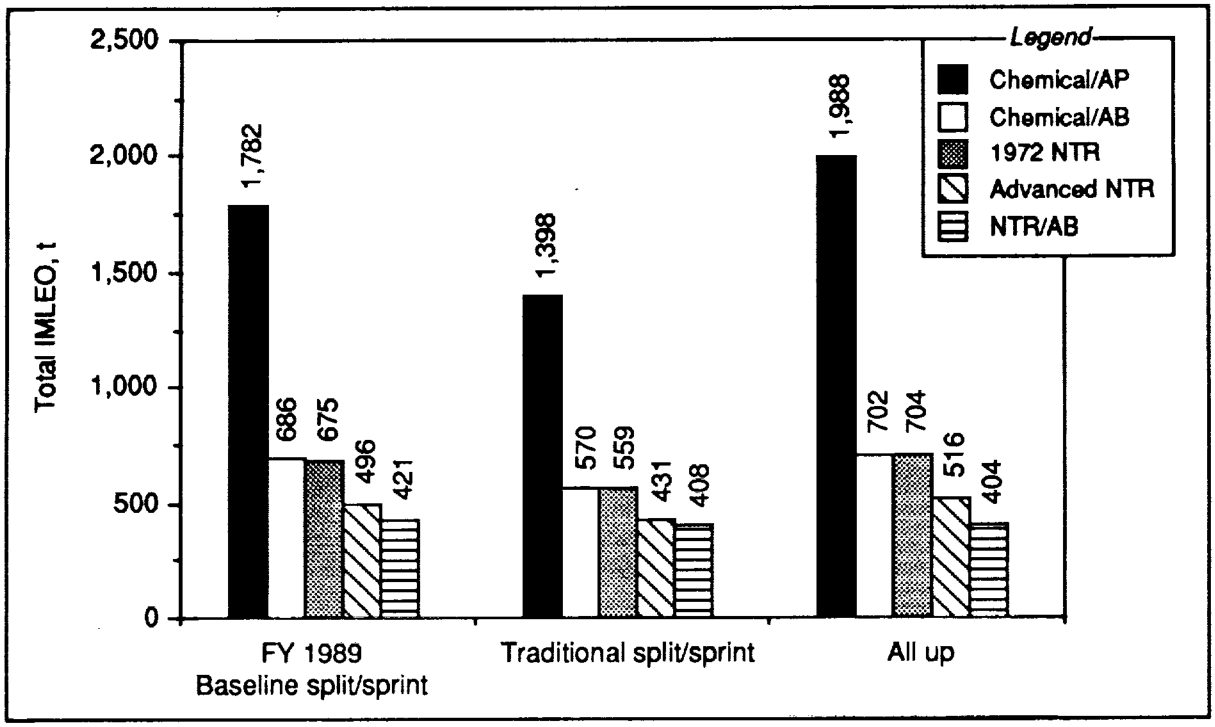


Figure 6.2.1-3.- 2002 Mars Expedition case study – IMLEO sensitivity to mission flight mode.

propulsion system performance, the chemical/AB and 1972 NTR systems have comparable IMLEO, with the advanced NTR and NTR/AB showing continuing reductions in LEO starting mass. Only results for the 1972 NTR and advanced NTR are presented for the all-propulsive NTR systems, because they are expected to bracket performance characteristics for the NTR.

6.2.1.3 Conclusions

Comparing chemical/aerobrake and all-propulsive NTR systems indicates that both concepts can accomplish the various types of missions while remaining within the 2-year, 1,140 t mass limit. The IMLEO results were shown to strongly depend on mission opportunity, flight mode, and the assumed value for the aerobrake mass fraction.

Due to the important mission improvement that may be possible using NTR and aerobraked systems, continued evaluation of both chemical/aerobrake and all NTR systems in the FY 1990 studies would be appropriate. Specifically, consideration should be given to (1) the characterization of a state-of-the-art NTR propulsion system point design that could support Mars missions in the 2000 to 2010 time frame, (2) continued aerobrake point designs to more firmly establish aerobrake configurations and mass fractions and to identify the impact of aerobrake usage on the overall vehicle design, and (3) continuation of mission mode studies that take advantage of the characteristics of each propulsion system.

6.2.2 Future Propulsion Technology Study

To assess advanced propulsion concept alternatives to chemical rockets, nuclear electric propulsion, and nuclear thermal rockets, a representative sampling of other advanced propulsion concepts was studied: solar sails, solar thermal rockets, laser thermal rockets, laser electric propulsion, mass drivers, railguns, tethers, magnetic sails, and ultra high power nuclear and solar electric propulsion.

Most of these concepts operate in a high specific impulse, relatively low-thrust regime; therefore, an unpiloted Mars cargo vehicle was chosen as the reference case in which these concepts might be most useful due to a less restrictive emphasis on trip time. The reference NEP cargo vehicle mission from the Mars Evolution case study was taken as the baseline mission. Results for two NEP cargo cases, a baseline round trip and a one-way trip are shown in table 6.2.2-I.

TABLE 6.2.2-I.- NEP CARGO MISSION RESULTS

	Round trip	One way
IMLEO (t)	835	702
Earth-Mars trip time (days)	937	823
Total trip time (days)	1,323	—

These propulsion technology concepts are assumed to be implemented in the same 2014 time frame as the nuclear electric propulsion and nuclear thermal rocket technologies. For this assessment, cargo payloads were set at 400 t, based on the assumption that the Mars outpost is at that time becoming operational and requiring significant cargo deliveries. Because of the unique characteristics of some of these concepts, the payload was divided into smaller units (20 t minimum), thus requiring multiple vehicles to deliver the entire 400 t payload.

In all cases, the reference start orbit is LEO (500 km, 28.5 degrees inclination), and the destination orbit at Mars is 6,000 km (Phobos orbit). Some systems, such as solar and magnetic sails, cannot start in LEO, and require an additional form of propulsion to raise the vehicle to a suitable starting orbit. In other cases, such as laser thermal or laser electric propulsion, a second propulsion system is required to perform Mars orbital capture. In all these cases, the mass penalty of the needed auxiliary propulsion system is included.

Figures of merit for this study are IMLEO and trip time. For cargo applications, IMLEO is the primary parameter with trip time serving as a secondary discriminator. Trip time is identified as the Earth to Mars travel time, although the possibility of vehicle reuse was addressed.

Each propulsion system was assessed in terms of the common mission requirements described previously. Particular requirements of a given technology were then assessed to derive a reasonable mission scenario for each system, dealing with issues of deployment altitudes, infrastructure requirements, and mission staging that are unique to some technologies. Following development of the scenarios, Mars cargo vehicle concepts were developed using each of the candidate technologies. Because most of these propulsion concepts provide low thrust, the cargo vehicle mission performance (IMLEO, trip time) was obtained using low thrust trajectory codes.

Each system is summarized below, describing technology/performance assumptions, necessary changes in the reference scenario, and the resulting cargo vehicle performance in terms of IMLEO and trip time. Volume VI of this report discusses the study results in more detail.

Solar Sails. A range of materials and construction techniques was assumed, from a near-term, deployable system to advanced sail designs requiring orbital construction and assembly in LEO. All concepts are restricted to orbits greater than 2,000 km due to atmospheric drag. Varying numbers of sails, up to 20, were considered to carry the payload; the greater the number of sails, the shorter the trip time and the greater the initial mass. A comparison of the shortest trip time case for two different size Drexler sails, a Staehle sail, and a Garvey sail is shown in figure 6.2.2-1. In all cases, either chemical or SEP OTVs were assumed for transport to 2,000-km orbits. Earth-Mars trip times range from 471 to 1,623 days, and initial masses range from 700 to 950 t. This performance is comparable to NEP system performance, although requiring an additional 100 to 200 days in trip time, and it represents a significant reduction in IMLEO, although 20 vehicles are required to achieve these results.

Solar Thermal. High performance solar thermal propulsion, using hydrogen propellant heated by solar energy, is capable of 1,200 seconds Isp with system thrust/weight ratio levels higher than most electric propulsion systems. The thrust/weight ratio was also regulated by varying the number of vehicles and system solar power levels. In all cases in which solar thermal vehicles were allowed to spiral out from LEO to Earth escape, the missions were found to have a minimum initial mass of 6,000 t IMLEO for Earth - Mars trip times ranging from 400 to 650 days (figure 6.2.2-2). This is almost four times the mass of a chemical cargo vehicle with aerobrake; therefore, the use of solar thermal propulsion in this mode is an unacceptable form of advanced propulsion for Mars cargo missions. The inferior performance results from the relatively low Isp, which is not sufficient to overcome the g-losses inherent in the low-thrust trajectory required. Propellant mass could be reduced to levels comparable to or less than the reference chemical/aerobrake mission by using the solar thermal rocket as an OTV for the Earth escape and by thrusting only near perigee; this approach has been shown to utilize propellant more effectively in chemical and nuclear rockets. Further reduction can be obtained by using a chemical kick stage just prior to escape and an aerobrake at Mars. Independent analysis by Martin Marietta has indicated initial masses on the order of 2/3 the chemical vehicle mass for solar powers on the order of 20 MW with a corresponding 15 percent increase in trip time over the chemical/aerobrake mission. Further study with comparable ground rules is required to determine the quantitative benefits of this approach. In addition, solar thermal propulsion may be used for Earth orbital transfer and perhaps Earth-Moon transport, where the required velocity increments are correspondingly lower.

Beamed Power Electrical/Thermal. The use of an external beamed power source to provide thermal or photovoltaic power to a cargo vehicle allows the use of relatively lightweight onboard receivers rather than a complete power system, with a commensurate improvement in vehicle thrust/weight. Laser and microwave power were considered over a range from 1 to 30 MW beam power. Thermal propulsion Isp's were assumed to be 1,500 sec; electric propulsion Isp was 5,000 sec. Because of diffraction limits in optics size and beam propagation, the beamed systems were not capable of operating effec-

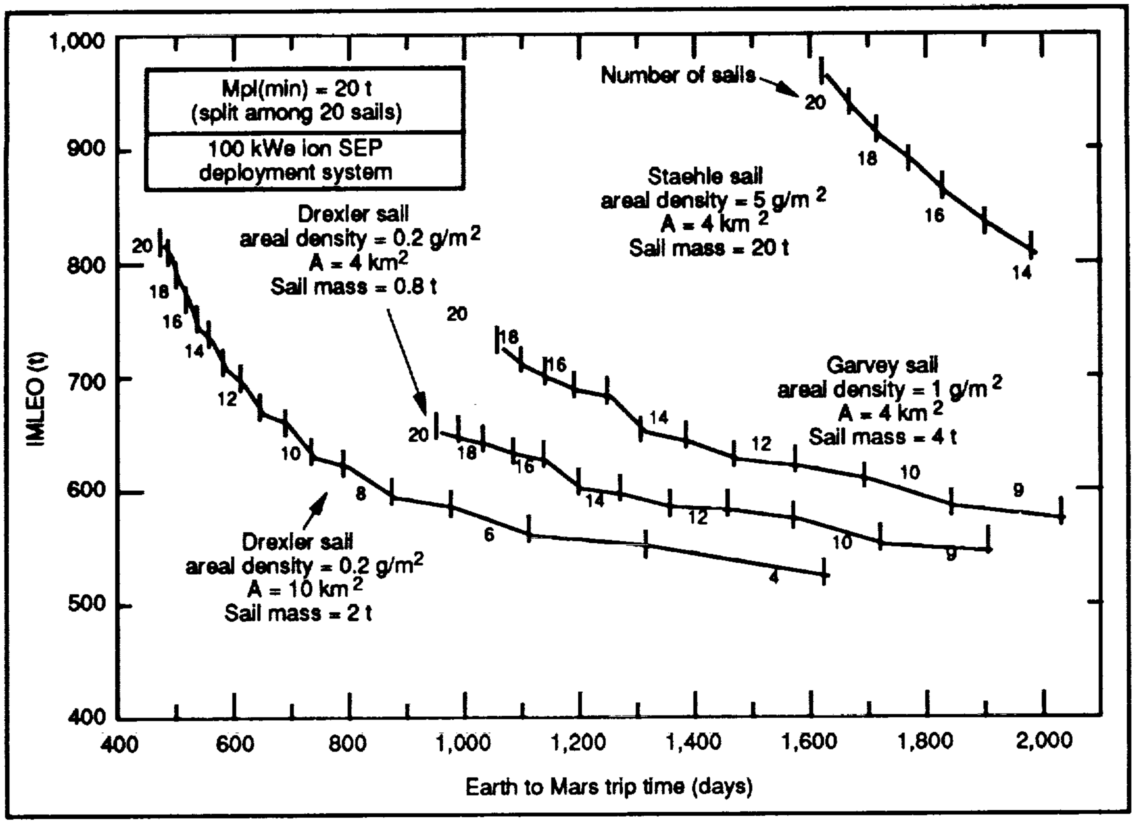


Figure 6.2.2-1.- Solar sail performance for the Mars cargo mission.

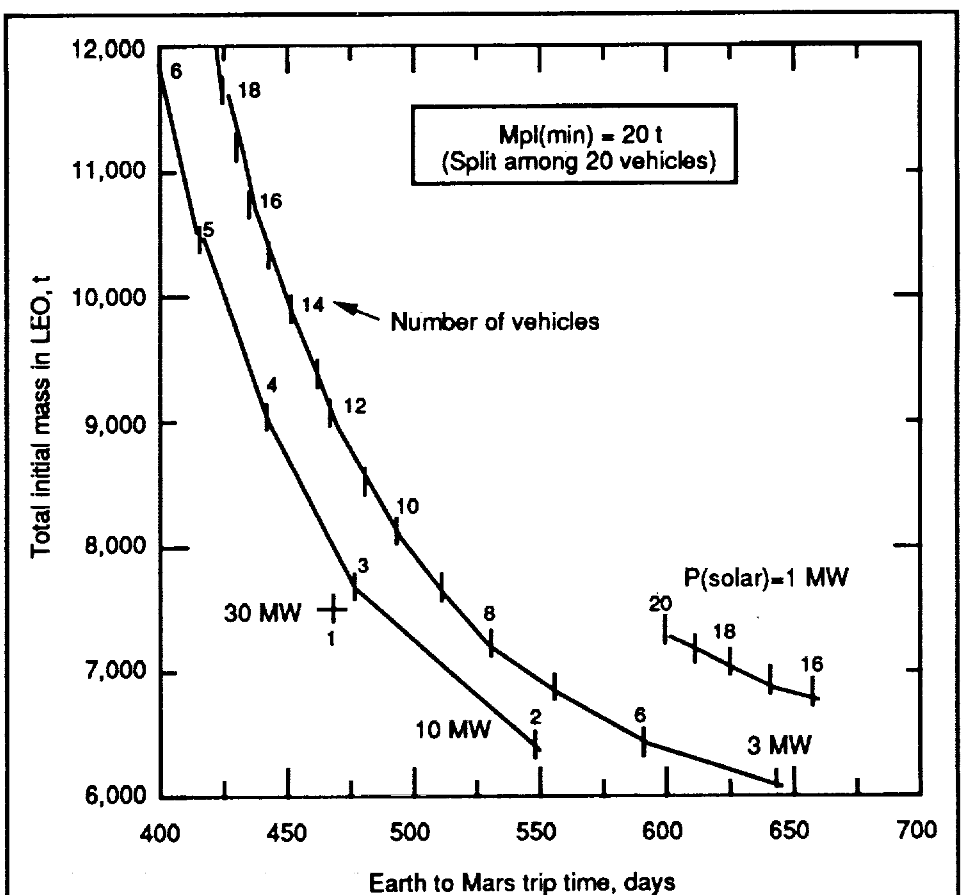


Figure 6.2.2-2.- Solar thermal performance for the Mars cargo mission.

tively beyond GEO. Therefore, the mission scenario used the beamed propulsion to raise the spacecraft from LEO to GEO, and chemical propulsion and aerobraking were used for the rest of the mission. The laser thermal vehicle was assumed to be expendable, and all other beamed concepts were assumed to be reusable. The laser electric system has a unique scenario, since it can generate its own power from solar radiation, and so it can function as a solar electric propulsion vehicle. Beamed propulsion provides a slight reduction in IMLEO of 1,000 to 1,500 t at the expense of increased trip time. Trip times of 300 to 500 days require 10 MW of beam power. Laser electric propulsion allows a reduction in IMLEO to approximately 1,000 t, again by increasing trip time to 600 days or more. The laser electric option also requires 10 to 30 MW of beam power. These systems in general are capable of only modest benefits, even when the laser infrastructure is not accounted for.

Mass Drivers and Railguns. These are forms of electric propulsion in which electromagnetic fields are used to accelerate projectiles or plasmas to high velocities to produce thrust. In a projectile mode, these devices are especially suited to in situ propellant utilization, since the acceleration process and propellant type are decoupled. For this study, the propulsion systems are powered by 1 to 50 MWe of power provided by a solar array. The systems were sized to achieve 1,200 seconds Isp. Solid oxygen (stored as a liquid and frozen into a projectile form prior to launch) was considered as propellant. Oxygen is a potential lunar resource for propellant production, and because it ultimately sublimes from solid to gas, there is no lasting projectile hazard.

Because of its low efficiency (less than 50 percent), the railgun was not competitive with the chemical vehicle. Mass drivers show some benefit over the chemical vehicle, provided that lunar oxygen is being produced and transported to LEO for other reasons. These concepts are comparable to the NEP vehicle; however, they also require MWe power levels and advanced solar cell technology. Without oxygen availability, neither concept is remotely competitive with either the chemical or NEP vehicles.

Tethers. By means of momentum transfer, tethers can transfer momentum to add velocity to one vehicle at the expense of another. Tether length, strength, density, and the masses of the two vehicles at either end all influence the performance of such a system. For this study, tethers were used in Earth orbit to provide some velocity prior to the chemical trans-Mars injection, and at Deimos and Phobos to perform orbit capture and transfer to Phobos without propulsion. Tethers at Deimos and Phobos allow the spacecraft to go from Mars escape velocity to Phobos orbit with minimal propulsion requirements. This use of tethers at Phobos and Deimos requires a great deal of initial infrastructure to deliver the tether facilities. In addition, the orbital requirements of the LEO assembly node require 20 cargo vehicles. These two factors combine to produce an unacceptable IMLEO for use of tethers for the cargo vehicle.

Magnetic Sails. The magnetic sail, or magsail, is a superconducting current loop that creates a magnetic dipole field that deflects the charged particles of the solar wind and obtains thrust. Because of its magnetic nature, the magsail must be operated in heliocentric space, outside the influence of Earth's magnetic field. The scenario assumed for the Mars cargo mission consists of vehicle and payload transport to Earth escape (1 au solar orbit) via SEP OTVs, at which point the vehicle and payload are assembled and deployed. The sail then follows a minimum energy transfer orbit to Mars. The magsail is incapable of circularizing at Mars orbit; instead, an aerobraked delivery of the payload to Phobos orbit is assumed. The sail then returns to a 1 au orbit at Earth. Both magsail trip time and IMLEO are comparable if not inferior to chemical/aerobrake cargo vehicle performance, and this concept does not show significant benefit for a Mars cargo mission.

Ultra-High Power Electric Propulsion. Due to the economies of scale, using hundreds of MWe of solar or nuclear electric power provides the potential benefit of high thrust and low specific mass propulsion systems. Both solar and nuclear power systems were considered for this application. A proposed nuclear system has a specific mass less than 1 kg/kWe using advanced reactor and dynamic power conversion concepts. The solar power system specific mass is 3.6 kg/kWe; there is no economy of scale in this case, but there is a gain in performance through advanced high efficiency, radiation-resistant solar cells.

Both systems depart from Earth orbit, travel to Mars/Phobos orbit, and return. Both concepts provide IMLEO levels comparable or superior to the reference chemical system, as shown in figure 6.2.2-3. The quick trip times of the ultra-high power systems indicate a greater possible benefit for piloted missions rather than for cargo missions.

Assessment. Of the concepts studied for a Mars cargo mission, solar sails and ultra-high power electric propulsion show the greatest benefit in payload delivery. The sails accomplished a Mars cargo mission with trip times and IMLEO characteristics only slightly inferior to a reference NEP cargo vehicle. The 100 MWe class EP systems outclassed the reference cargo vehicle in terms of trip time, with comparable IMLEO; greater benefit would be seen in using such systems as piloted vehicles. Of the remaining concepts, most provided little or no benefit over a chemical/aerobrake system, much less a

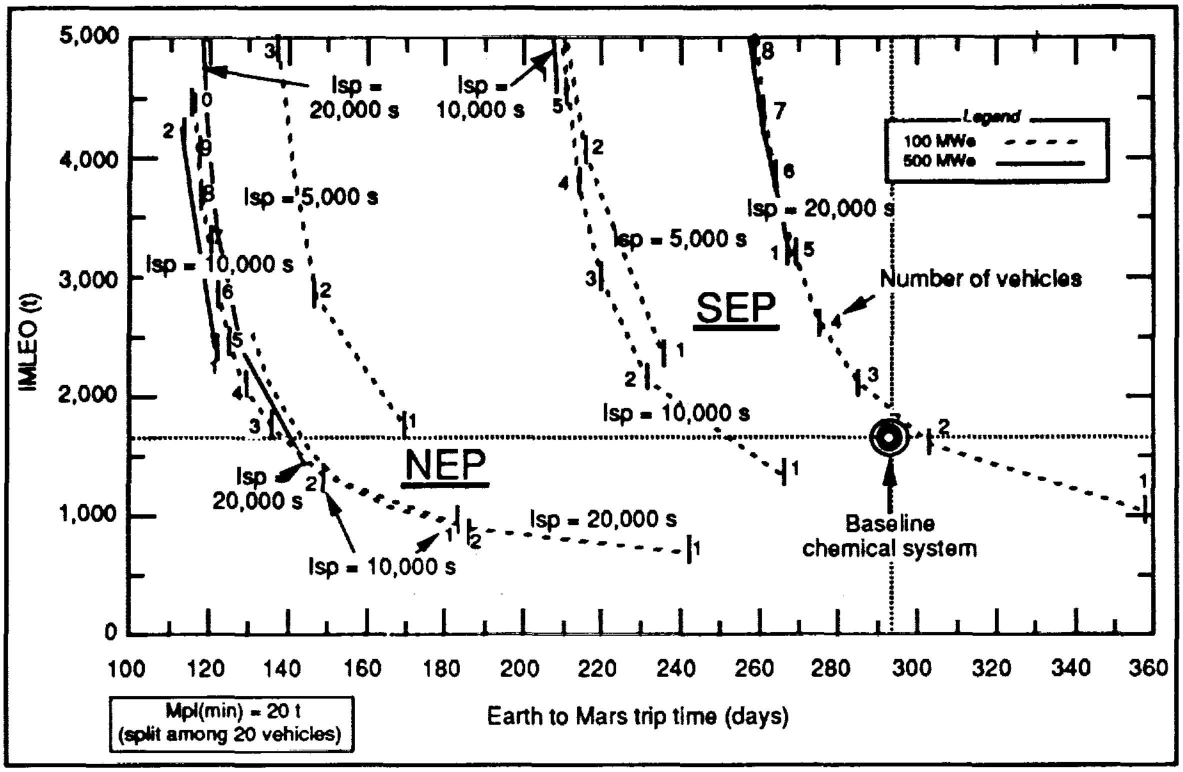


Figure 6.2.2-3.- Ultra-high-power electric propulsion performance for the Mars cargo mission.

NEP cargo vehicle. Figure 6.2.2-4 shows representative values of IMLEO and trip time for each system in comparison to both the chemical and baseline NEP cargo vehicle results, indicating the relative merit of each.

Application of all these systems to another mission could yield different conclusions because of different mission trajectory requirements or emphasis on figures of merit other than IMLEO. Furthermore, the mission being studied could be tailored for each propulsion system to best utilize a particular technology. (Some of this tailoring was necessary in the course of this study.)

6.2.3 Fuel Systems Architecture Assessment

The objective of this multi-year study is to define management architecture options for launching, storing, and transferring cryogenic fluids throughout the Earth-Moon-Mars system in support of the node and transfer vehicle concept definitions.

Cryogenic propellant management spans all the case studies; therefore, it is subject of a broad trade study. The study was performed by defining a trade space (figure 6.2.3-1) to develop fuel system architecture options based on case study requirements.

In FY 1989, it became clear that assessing the entire trade space would be extremely time-consuming and top-level, given time and manpower constraints. A focused approach provided depth on a few representative concepts that could provide insight for future trade space reduction or emphasis. It was decided, somewhat arbitrarily, to concentrate on defining free-flying concepts for LEO (keeping propellant away from human activities), using the largest ETO vehicle option (Shuttle-Z) and the propellant requirements of the Mars Evolution case study, with additional requirements for unmanned missions (GEO, planetary, etc.). The techniques for transfer would involve a zero gravity, thrust settling assisted transfer to permit a common design for comparison.

6.2.3.1 Lunar Evolution Case Study Focus

The impact of Lunar Evolution case study constraints and mission drivers was assessed at a top level. The major drivers on the fuel system architecture are:

a. All space vehicles are reusable. Liquid oxygen/liquid hydrogen propulsion is used for all transfer and ascent/descent vehicles.

b. ETO vehicles must support propellant exchange/transfer.

c. No orbital nodes other than Space Station Freedom are used.

d. Space Station Freedom will not be used to store main-stage propellants.

e. Lunar liquid oxygen is one-half of that needed for round-trip of cargo or crew.

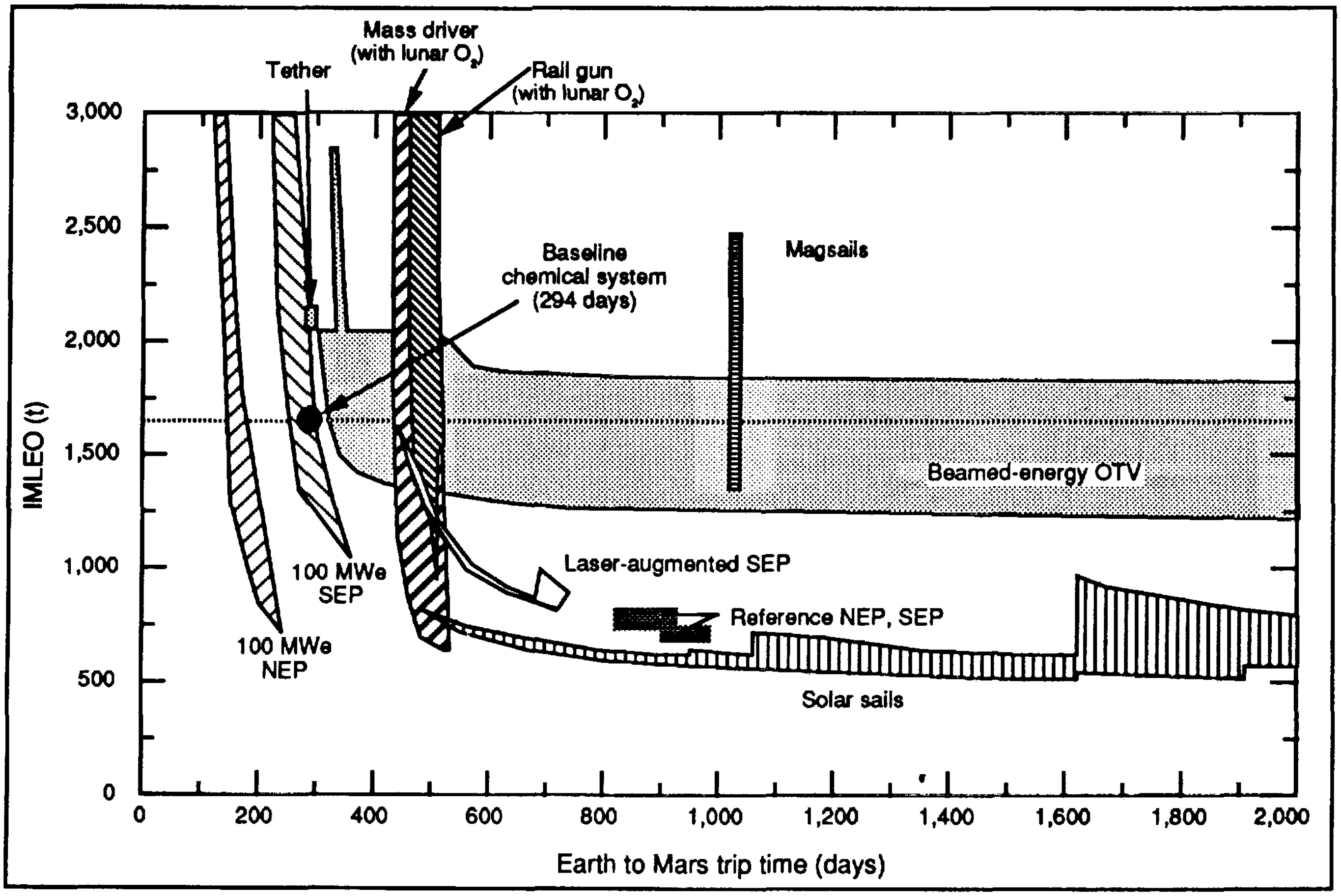


Figure 6.2.2-4.- Summary of future propulsion performance for Mars cargo mission.

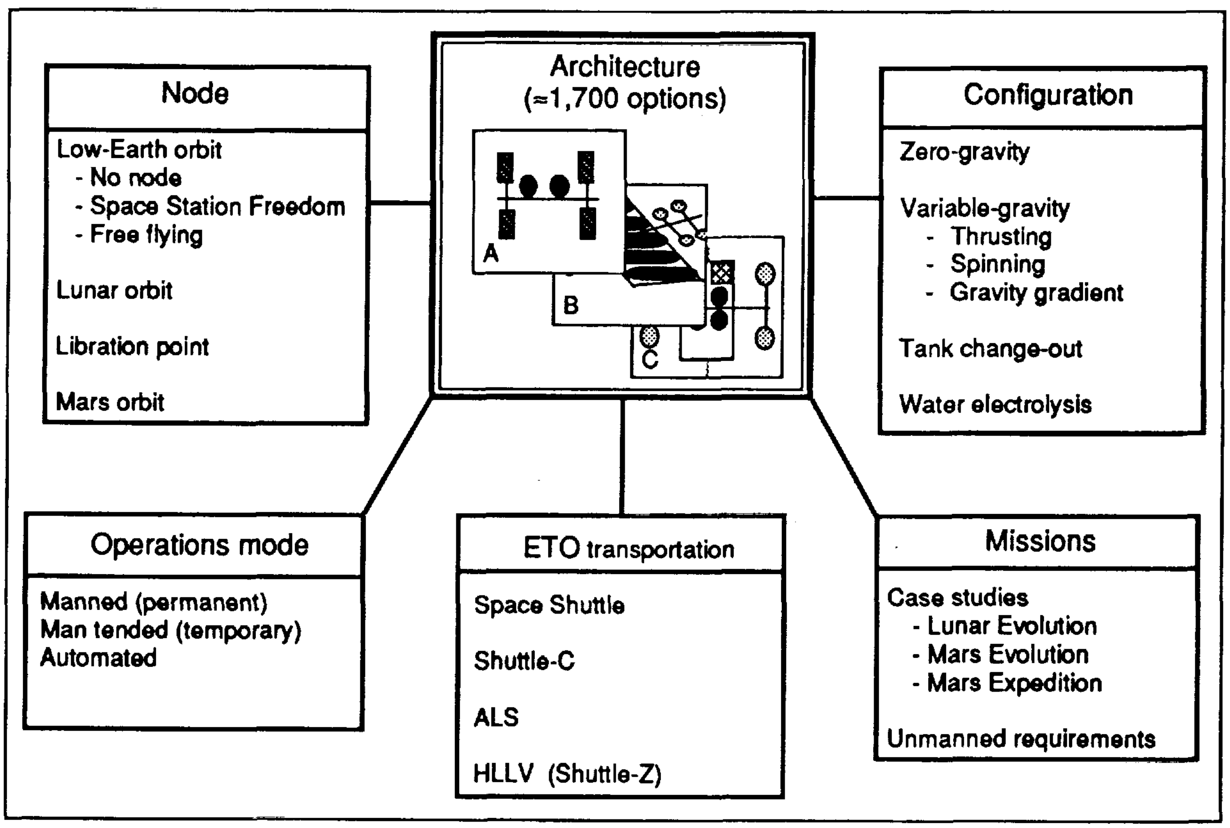


Figure 6.2.3-1.- Defined trade space for fuel systems architecture assessment.

Based on the mission drivers and the defined LTV configurations, three options were assessed: (1) tanker resupply for each mission, (2) temporary depot based on Shuttle-Z third stage, or (3) a permanent free-flying depot structure. The permanent depot was preferred but violated the case study ground rules. The temporary depot provides similar advantages (low boiloff, minimum ETO launches, and ETO schedule independence) with the addition of commonality for a Mars mission. However, a temporary depot requires the development of a heavy-lift launch vehicle.

Two options for transporting hydrogen fuel to combust with lunar oxygen were assessed: (1) transporting spherical tanks as cargo to lunar orbit during the cargo missions (manifest of cryogenic propellant with the crew provides limited capability and is an unnecessary crew hazard), and (2) increasing the hydrogen tank volume to manifest the propellant cargo with the mission propellant. The first option requires 14 t of hydrogen on the cargo mission with tanks ranging from a single 7.3 meter diameter tank to four 4.6 meter diameter tanks. The second option requires 7 t per mission (cargo and piloted) within the vehicle's tanks with only a 1-meter increase in diameter. The preferred method is to increase transfer vehicle tankage and transfer propellant in lunar orbit.

6.2.3.2 Mars Evolution Case Study Focus

Various trades were performed to determine the most efficient and least complex concept for propellant management. Some of the key trades involved assessments of the structural configuration, storage tank configurations, pressure system for propellant transfer operations, and debris protection.

The structural configuration options consisted of the 5-meter truss planned for Freedom, a large box truss that would be tank configuration specific, and a tetrahedron structure. The tetrahedron structure offered the lowest mass due to its inherent characteristics, and it provided an opportunity to minimize assembly EVA.

The storage tank configuration options consisted of single, very large cylindrical tanks, common spherical tanks, and Space Shuttle sized cylindrical tank sets previously defined by General Dynamics during the Long Term Cryogenic Storage Facility Study performed for MSFC. The common spherical tank was chosen because of its low boiloff, structural configuration compatibility, lowest complexity on-orbit assembly operations, and ground handling implications.

Pressure for expulsion of propellant can be obtained by various methods, such as storage of high pressure helium, but due to the large quantities of propellant that must be transferred, the method must be efficient and reliable without introducing unwarranted complications and safety hazards. All options addressed used the propellant gases to minimize logistic requirements and high-pressure storage requirements. The options consisted of real-time (produced when needed) gas generation using heating elements to force gas production, a stored boiloff system using conditioned stored boiloff, and a turbo pump using a gas generator to drive the pump. Based on determination of the energy consumption, systems required, weight, complexity, safety, and overall efficiency, the real-time pressure generation system was chosen.

With continued growth in man-made orbital debris, debris protection is an increasingly important design consideration for low-Earth orbit spacecraft. A dual bumper shield was investigated for both the entire space structure and individual tank and line protection. There was no major difference between the two methods based on weight, but the common shield required assembly EVA or complicated deployment techniques. The individual system protection was chosen for this concept.

The concept developed to support the Mars Evolution case study that could perform zero gravity or thrust induced gravity (for propellant settling) transfer is shown, along with its characteristics, in figure 6.2.3-2. The depot concept consists of four regular tetrahedron substructures (one oxygen, three hydrogen), which include the spherical propellant storage tank, high-pressure storage tank, pressure generation system, and a service substructure. The service substructure contains a guidance, navigation, and control system, a photovoltaic array (7 kW), batteries, fuel cells, GN&C, refueling nozzles, and docking provisions consisting of two attach points at the refueling nozzles and a strongback to accept a third attach point along its length to permit different vehicle sizes. The concept uses gaseous hydrogen/oxygen attitude control and reboost engines, which also provide thrust-induced gravity to take advantage of the cryogenic propellant boiloff and eliminate the need for a separate propulsion propellant (such as hydrazine) that must be replenished.

Each tetrahedron substructure would be launched with the storage tank integrated with its associated thermal insulation (120 layers), debris shield protection, plumbing, and pressurization tankage. The tetrahedron structure provides the support required for launch based on STS launch load conditions. The hydrogen storage tanks are launched full. The oxygen storage tank is launched empty. Liquid oxygen supply tankers are launched as an integral element of each assembly launch manifest. The total depot system, including propellant, is launched in six Shuttle-Z-equivalent flights. The substructures are attached with the use of corner guides and a power

Dry Mass	**44,500 kg**
Propellant Type	**LOX/LH 2**
Mixture Ratio	**6:1**
Storage Tanks	
Hydrogen	
Number	**3**
Size (spherical)	**10.5 m Diameter**
Propellant Capacity	**117,300 kg**
Oxygen	
Number	**1**
Size (spherical)	**10.5 m Diameter**
Propellant Capacity	**625,000 kg**
Total Mass	**792,000 kg**

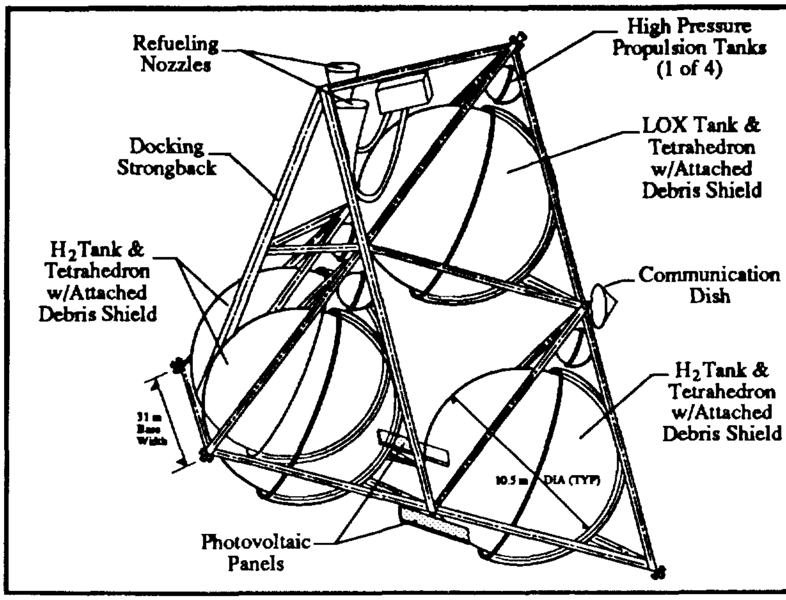


Figure 6.2.3-2.- Preferred configuration for reduced matrix.

winch to reduce the EVA required for assembly. The structure requires temporary struts for stabilization during assembly.

The depot has three operational phases: (1) depot resupply, (2) STV fueling, and (3) storage. The operational procedures for each phase were assessed. Depot and STV fueling resupply operations are very similar, with the depot providing all services (e.g., pressure for tanker to transfer propellant to depot). An initial evaluation of the transfer operation indicated that propellant fluid motion and large center-of-gravity shifts required active control throughout the procedure. Thrusting through the reboost engine to provide this control was also sufficient for propellant settling (10^{-4} g's) and provided part of the reboost energy required. During the storage phase, a thermodynamic vent system is required to maintain tank pressure without expelling liquid. A zero- or low-gravity liquid acquisition device is required for the vent system to provide given initial conditions to the system and avoid complex control systems. Settling of the propellant by thrusting to develop a defined liquid/gas interface for pressure control by gas expulsion consumes enormous amounts of propellant, making this method impractical.

Definition of the subsystems and the operational procedures for the resupply of the depot and an STV allowed the determination of the propellant utilization for these operations. Volume VI provides a summary of the propellant losses during the various operational phases of the depot for the first series after assembly. The largest loss is due to operational losses associated with attitude control and reboost of the facility. Alternative configurations and methods of providing gravity gradient stabilization (to minimize attitude control propellant expense) will be investigated in future studies.

Cryogenic fluid management encompasses a large number of new technologies that are essential for successful depot operations. These technologies can be divided into four major areas: (1) storage, (2) supply, (3) transfer, and (4) fluid handling.

Long-term storage of cryogenic fuels is essential for minimizing propellant losses between missions. Confident system design criteria must be established for the development of future cryogenic storage thermal systems to provide low-conductivity structural supports, high-efficiency multi-layer thermal insulation protection, minimum system weight, understanding of system integration (thermal/structural), high reliability, and repeatable fabrication techniques. Methods must be developed to control storage system operating conditions, minimize temperature stratification of the fluid, and avoid unpredictable pressure surges. The effects of the launch environment (vibration, acceleration, pressure differential), the space environment (debris, micrometeorites, atomic oxygen), the degradation due to prelaunch purge systems, and ground handling of the thermodynamic protection system must be understood.

The capability to handle large quantities of cryogenic fuels safely in a low-gravity environment without causing major dynamic control problems to the overall system is essential. Predicting fluid motion is critical to the development of an attitude control system for a depot. To do this, gravitational environment effects, flow in-

duced sloshing, the need for and impact of baffling, and the impact of center of gravity shifts resulting from liquid transfer operations must all be understood.

Supplying cryogenic fuels safely and efficiently in a zero- or low-gravity environment is one of the most difficult but essential capabilities for performing transfer operations in space. Methods must be developed to provide required pressure differentials and liquid subcooling with minimum power and propellant losses. An understanding must be established of the pressurant temperature effects: pump/compressor system complexity, reliability, and efficiency (minimizing heat addition to transferred fluid); pump cavitation criteria; and acquisition/pump interactions. The fluid acquisition method must be efficient, create minimal thermal disturbances, meet outflow demands, and minimize residual propellants.

The capability to transfer cryogenic fuels safely and efficiently in a low-gravity environment to supply fuel depots and fill vehicle fuel tanks will also be essential. The technology will enable single phase liquid transfer with minimal propellant losses during transfer line and tank chilldown, determine the proper liquid injection technique and sequence into the receiver tank, understand the effects of low gravity or acceleration environment on transfer operations, prevent inadvertent venting of propellant during the fill process, provide accurate mass flow measurements, and most important, provide predictable tank filling capability. The proper liquid injection technique and sequence into the receiver tank must be determined, and the effects of low-gravity or acceleration environment on transfer operations must be understood.

Upon completion of the evaluation of this preliminary depot concept for LEO, some observations can be made:

a. Boiloff losses for storage of propellants are not the only source of concern. The propellant losses due to operations for stability and reboost consume a considerable amount, as well as the transfer losses for the depot itself.

b. Boiloff attitude control, pressurant, and depot reboosting can provide a useful and essential capability. All the propellant can be used without a need for unused boiloff losses. Reliquefaction of boiloff is unnecessary and only complicates the overall system.

c. Power requirements can be much higher during the transfer operations, especially if the transfer operation must be accomplished quickly.

d. Thrusting may be required during transfer operations to maintain control of the vehicle/depot configuration. Thrusting during transfer can be adequate for propellant settling at the sump, but a low-gravity liquid acquisition device will still be required for storage operations.

6.2.3.3 Mars Expedition Case Study Focus

The impact of Mars Expedition case study constraints and mission drivers were assessed at a top level. The major drivers on the fuel system architecture are:

a. All space vehicles are expendable.

b. The ETO transportation must support propellant exchange/transfer.

c. No orbital nodes are required.

d. Design for 1995 technology.

An assessment of the mission drivers and STV configurations based on the Shuttle-Z showed that the trans-Mars injection stage, also the Shuttle-Z's third stage, should be launched with a refueling tanker. The refueling tanker would resupply the injection stage during third stage ascent to take advantage of the acceleration-induced gravity and to minimize tanker storage time. This minimizes ETO launches and transfer losses, but at the cost of increased dry mass and boiloff (which could be used as described previously).

The propellant tanks for the return to Earth and Mars operations must be fueled on-orbit to meet manifest and design constraints. Propellant transfer capability (pumps, pressurization, liquid acquisition in zero gravity, and refueling couplings) was added to a TMIS to perform this operation and to replace the propellant that boils off in the TMI stages after launch.

According to the current Pathfinder schedule, the technologies required for TMIS propellant storage, transfer, and management will not be available by 1995. The flight experiment program must be accelerated; otherwise, alternative methods must be used, or higher risk must be incurred.

6.2.3.4 Summary

Past efforts have provided insight into various aspects for defining the fuel system architecture that may be required for future human exploration missions. In particular:

a. Propellant losses are not limited to boiloff and must be factored into future designs to minimize all losses.

b. Operational losses (attitude control and reboost) can have a significant impact on propellant losses for on-orbit storage of large quantities of propellant.

c. Boiloff losses can and should be used to the depot's advantage, eliminating the need for reliquefaction systems.

d. Required transfer times can have a significant effect on surge power requirements.

e. Expansion of the transfer vehicle's tankage of propellant may be the most efficient method for propellant logistics to the Moon.

f. Fluid management technology will be required for the depot and transportation vehicles.

g. Low-gravity propellant settling may be the most attractive method for propellant transfer.

6.3 LIFE SUPPORT SYSTEMS SPECIAL ASSESSMENT

6.3.1 Life Support Architecture Study

This section reports on the examination of the impact of life support approaches to the OEXP case studies and identifies some of the initial results that have been obtained. The 2-year study is currently focused on developing comparative data for the application of several life support system approaches to a set of mission drivers. Those drivers reflect the number of crew, mission durations, and resource availability for space facilities that range from lunar transfer vehicles to long-term outposts on the surface of Mars. The comparative data for 2 to 30 crew and mission durations from 4 days to 12 years will be tabulated and discussed in a mission planner's life support system (LSS) guidebook. In addition to the mission impact parameters for the tables, the development of the guidebook will produce technology assessments and systems compatibility analyses, which will provide a guide to prioritizing future LSS development. The objectives of the study are:

a. Identify LSS evolutionary paths.

b. Minimize life support system logistics.

c. Integrate use of in situ resources.

d. Determine physical-chemical/biological crossover.

The tabulated parameters and technology assessment discussions of the guidebook focus on providing information to help satisfy these objectives.

6.3.1.1 Guidebook Description

Volume VI of the Exploration Studies Technical Report includes a preliminary version of the mission planner's LSS guidebook as an initial output of the comparative results from the study. The guidebook examines the impact of using each of four different life support system approaches with varying degrees of closure for lunar outpost missions. The four approaches are (1) an open loop approach with a regenerable CO_2 removal process, (2) a closed loop physical-chemical system such as that planned for Space Station Freedom, (3) an advancement on the Space Station Freedom approach to reduce the expendables required for the system, and (4) a biological system for nearly complete resource recycling and life support.

The mission drivers for the guidebook were defined to represent lunar outpost life support missions. The mission is characterized as follows:

a. An installation is designed for long-term occupancy by a human crew on the surface of the Moon.

b. Duration of the stay at a lunar outpost by the human crew is 1 Earth year or longer (surface installations for shorter periods of time will be addressed in next year's studies).

c. Number of crew ranges from 4 to 30.

d. The outpost will be visited once every 6 months, after the first visit in about 7 months, by a transport vehicle from Earth; this vehicle will have resupply capabilities for life support system consumables and expendables.

e. The outpost may benefit from by-products of in situ extraction of propellants from the lunar soil. These by-products would be oxygen and/or water.

The lunar outpost was selected for this initial comparison for the following reasons:

a. Life support for a lunar outpost is likely to operate over longer durations and support larger crews than other missions, therefore requiring more reliable and advanced technologies.

b. Lunar outpost life support is a good candidate for system evolution to reduce system-related resupply. This means that system upgrades to more advanced technologies are more feasible than with other potential missions.

c. In situ manufacturing plants (for propellant) are being conceptualized for the Moon, and life support resources from these in situ plants are considered likely motivators for system upgrades.

d. A lunar outpost has the potential of evolving to a hybrid biological/physical-chemical life support system. Because of that, lunar outposts represent good candidates for the use of advanced LSS technologies in that area.

The main feature of the guidebook is the set of tables, which list the values for mission impact parameters such as mass, power, and volume for each of the different LSS approaches. Each table addresses a specific set of

LSS mission drivers that are related to a particular characterization of a lunar outpost mission. The mission impact parameters presented in each guidebook table show the effect that LSS approach selection would have on the design of a particular mission. Figure 6.3.1-1 shows the initial mass versus resupplied mass constituents for LSS of a crew of eight for 2 years using three different approaches.

The lunar outpost missions for the guidebook are characterized by crew size that ranges from 4 to 30. The intermediate number is eight. The crew of four scenario would be for outposts that are considered transitional. These smaller outposts would probably use a single habitation module (perhaps a variation on the Space Station Freedom design). The layout for a crew of four would be one habitation module connecting a scientific laboratory module and a unit for surface operations. The total enclosed volume of the four-person outpost is estimated to be about 700 m^3. The case involving a crew of eight would entail an early operational outpost established for the construction phase. The crew of 30 would occupy an outpost that is operational and evolving toward a self-sustaining colony on the Moon and would encompass about 3,400 m^3.

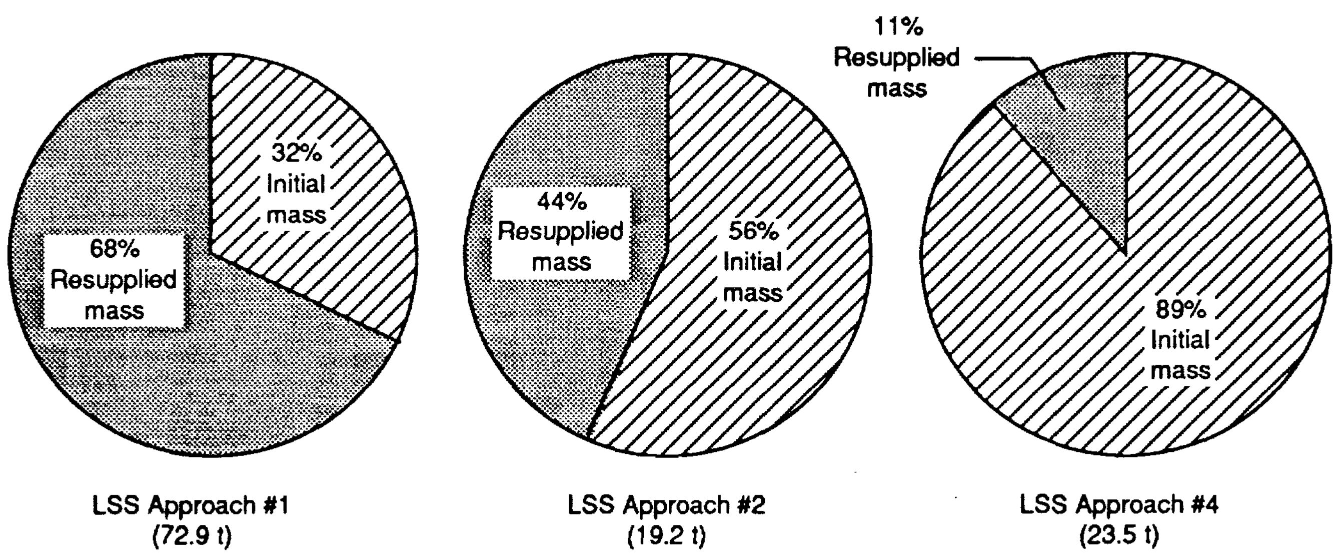
For the transitional and early operational outposts, the mission duration was considered to be 1 to 2 years. For shorter durations, the mission would be close to the early outpost category, and for longer durations it would tend toward operational status.

For the early operational outposts, the duration may be as short as 1 year, but more than likely 2 to 12 years would be considered. As the outpost evolves, the duration may go from 2 years to 12 years or longer.

Because the Moon is relatively close to Earth, every lunar mission will be serviced by resupply flights. For the development of the guidebook, it was assumed that for a 1-year mission duration, there would be one resupply, and a 12-year mission would be resupplied 23 times. Each resupply flight is assumed to have the capability to deliver the needed life support consumables, which will consist of food, nitrogen, oxygen, water, and the required LSS expendables, including spare parts. Figure 6.3.1-2 gives an example of the make-up LSS resources in a resupply delivery based on LSS Approach #2.

Two categories of in situ production were considered. The first category was for a transitory situation and would produce all the necessary oxygen for the life support after the in situ plant became operational, but it would not produce water. The second was the operational in situ production case, and it would manufacture all the oxygen and water needed for life support.

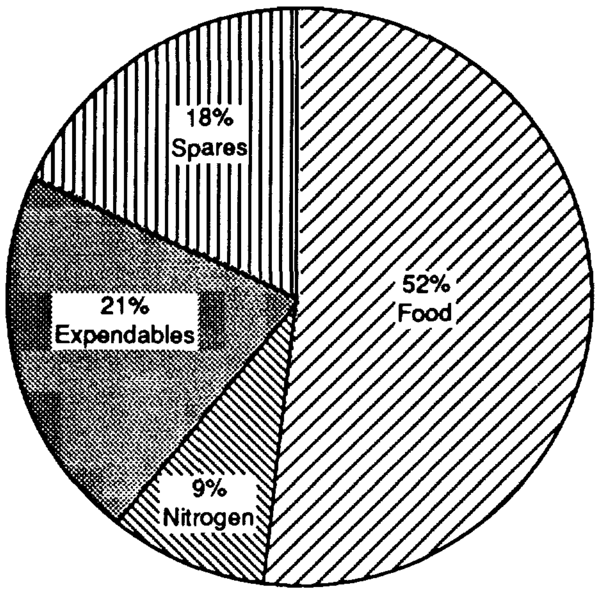
The other topic addressed briefly in the preliminary guidebook is an initial assessment of the technology issues related to the development of the LSS. Those



Notes:

1. Data based upon 4 crewmembers; 730-day mission
2. Initial mass includes LSS hardware and 190-day supply of consumables, expendables, and spares
3. Resupplied mass includes 540-day supply of consumables, expendables, and spares

Figure 6.3.1-1.- Life support approach – initial versus resupply mass.



Notes:

1. Data based upon 8 crewmembers; 180-day resupply interval
2. Nitrogen consumables include:
   127-kg module leakage
   303-kg two single module repressurizations
3. All oxygen and water are assumed to be recovered or regenerated

Figure 6.3.1-2.- Life support approach #2 – resupply mass breakdown.

technology issues are summarized as follows:

Water System

a. Simplify the Space Station Freedom water loops.

b. Eliminate urine pre-treatment chemicals.

c. Regenerate or eliminate post-treatment filtration and sorbent beds.

d. Develop reliable hardware with longer mean time before failure and shorter mean time to repair capabilities.

e. Improve process water quality monitoring

f. Develop automatic specific chemical compound identification for impurities.

Air System

a. Develop technology for generating make-up N_2.

b. Reduce power consumption for vehicle/outpost temperature and humidity control.

c. Improve CO_2 by-products technology.

d. Improve trace contaminant control and monitoring.

Waste System

Develop technology for converting LSS and crew waste into water, useful hydrocarbons, etc.

Food System

a. Reduce thermal storage requirements (refrigerator/freezer).

b. Integrate food production into the LSS.

These evolving technology topics will be examined during the continuation of the study. So far, the study indicates that technology advancement emphasis is moving away from reducing weight to technologies providing additional reliability through reduced complexity.

6.3.1.2 Conclusions

For the four approaches considered in the study (1) an open loop approach with a regenerable CO_2 removal process, (2) a closed loop physical-chemical system such as that planned for Space Station Freedom, (3) an advancement on the Space Station Freedom approach to reduce the expendables required for the system, and (4) a biological system for nearly complete resource recycling and life support, the following conclusions can be drawn:

a. Approach #1 has very large resupply requirements unless in situ water is available (Figure 6.3.1-3 compares total mass requirements for open loop against closed loop systems). This comparison indicates that a closed loop (regenerative) system would be more suitable for long duration space missions like a lunar/Mars outpost or transit to Mars.

b. Approaches #2 and #3 do not benefit greatly from in situ production, unless system flexibility for life support system expansion is considered.

c. The mass penalty for the biological system (Approach #4) occurs early but does not increase appreciably over time. This indicates that biological systems are more suitable for long duration missions where there is greater payoff on the initial mass investment.

d. In situ oxygen does not significantly benefit any of the life support approaches studied, but in situ water provides a great benefit with Approach #1 and a sig-

nificant benefit in support of outpost expansion with Approach #4.

e. Approach #4 has a significant technology impact as well as a weight, power, and volume penalty, because it depends heavily on versatile robotics for planting, cultivating, and harvesting crops. This robotic impact trades against the use of a large amount of crew time for tending the biological part of the life support system. Because Approach #4 tends to be labor-intensive, it is more suitable for missions where the number of crew is large.

f. Availability of in situ resources allows for open-loop expansion of the outpost to accommodate larger crews if waste disposal is not a problem.

6.3.2 Advanced Mission EVA System Requirements Study

6.3.2.1 Solar Flare Shelter Trade Study

Solar flares are little-understood energetic events on the solar surface that can threaten lunar missions by providing up to 1,500 rem radiation doses. Forty minutes of warning are provided by prompt X-rays propagating at lightspeed in front of the slower protons making up the flare's radiation threat. A massive flare may occur once every 10 to 20 years with smaller life- or health-threatening flares occurring one or more times a year. Dosages of 118 rem are fatal to 10 percent of the population; 345 rem are fatal to 50 percent of human recipients. Shielding against the possibility of such events is thus mandatory since, for any sustained lunar surface activities, the chances of crew fatalities due to a solar flare rise to unacceptable levels without shielding.

Based on standard dosage limits, a shield of either 9 cm thick aluminum or 16 cm thick lunar regolith would protect astronauts on the Moon from all but the most energetic of solar flares. To handle the most energetic of flares, the shield thickness must rise dramatically because of secondary radiation effects. That is, a high-energy primary particle striking a thinner shield produces a shower of lower energy secondary particles, magnifying the damage produced. A shield composed of lunar regolith can be provided over the lunar habitat, and in case of a flare, the EVA crewmembers would have a minimum of 40 minutes to retire to safety. If the crewmembers are working at a site much more than 40 minutes of travel time away, however, they would not be

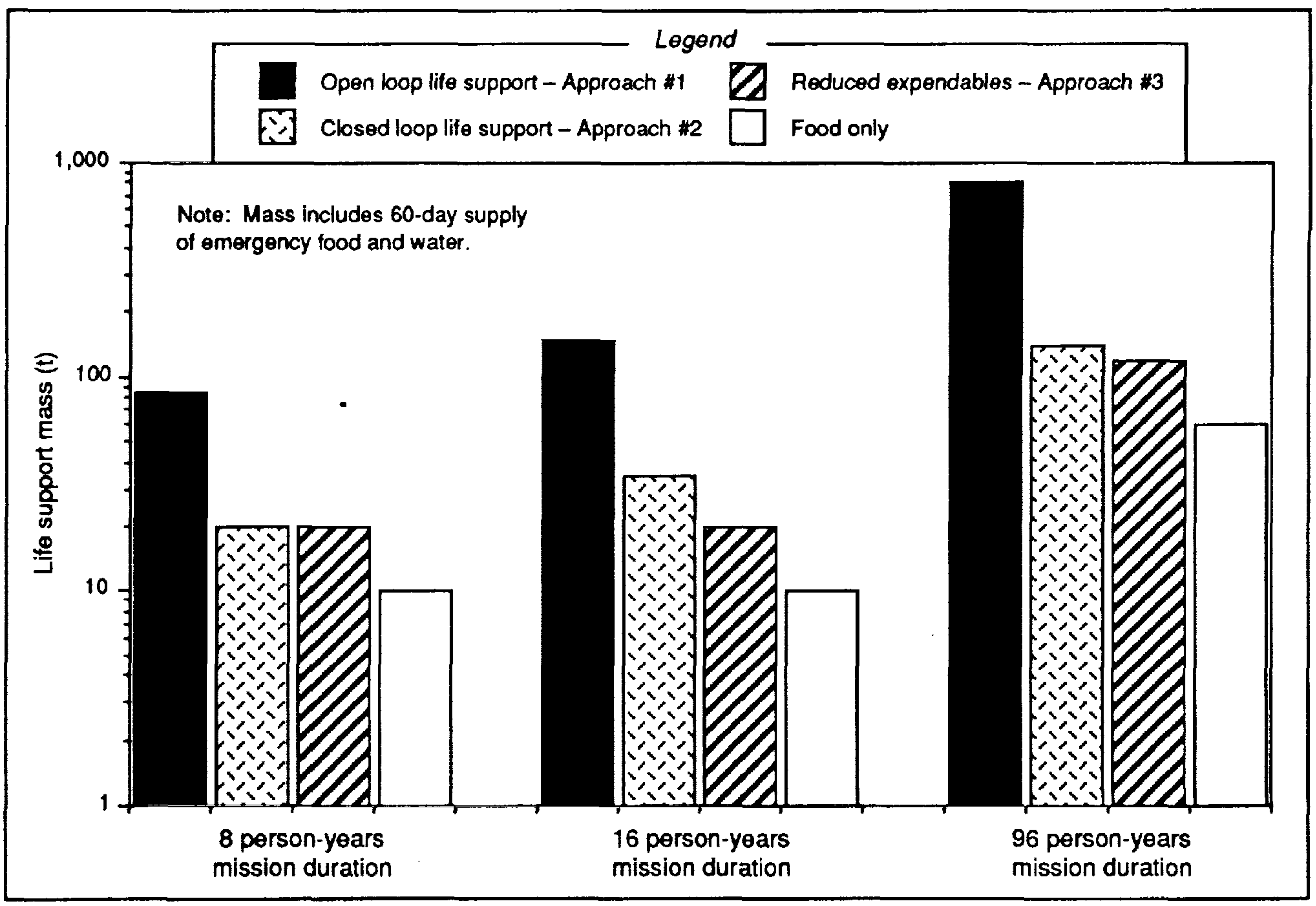


Figure 6.3.1-3.- Lunar outpost life support comparison.

able to count on the protection of the habitat. With current lunar rover technology, it seems reasonable to assume an upper limit of 10 km/hr on velocity. This means that any EVA crewmember working at a site more than about 10 km from the main habitat must be provided with some form of contingency solar flare shelter. Otherwise, the risk of death or injury from a flare is unacceptably great for routine lunar operations.

Three methods of providing the contingency shelter have been proposed. First, an expedient shelter based on a contingency nuclear fallout shelter could be constructed. In this scenario, explosives would be used to excavate a trench in the lunar surface 2 meters deep and 3 to 5 meters long. The rover would then be driven over the trench, and its equipment and structure, together with regolith piled over the rover, would provide a radiation-proof cover for the trench. The EVA crew would lie in the trench for the 8 hours to 4 days required for the flare to subside, drawing consumables from the rover through umbilicals. Unfortunately, a reasonably optimistic estimate of the time required to construct the shelter, taking into account site finding, explosive emplacement, trench preparation, and regolith moving, indicates that such construction would require on the order of 2 hours, which is far too long. Approximately 650 kg of equipment must be added to the rover to perform this operation. With the difficulties and uncertainties related to construction time, this approach does not seem feasible.

The second method of providing contingency flare sheltering would be to prepare in advance a network of distributed shelters so that any remote worksite would be within 10 km of at least one such shelter. These shelters would resemble the expedient shelter but would have roofs made by piling regolith over a kevlar "tarpaulin" pegged over the trench. The logistics and man-hours required to construct such a system would be prohibitive, even if it were constructed one additional shelter at a time as the outpost was expanded. Realistic problems with the construction technique could exacerbate the problem.

The third method of providing contingency flare protection would be to armor the rover vehicle so that in case of a flare, the crew could retire to its interior and drive safely back to the habitat. The armor could be provided by filling a double-walled crew cab with lunar regolith. Although such a shield and the rover suspension to carry it would increase the rover mass to 10,000 kg, only 20 horsepower would be required to drive a tracked vehicle this size up a 15-degree grade on the lunar surface. Two horsepower would be required for level cruising.

Although no method yet proposed is entirely satisfactory, the armored rover vehicle approach to solar flare sheltering has the fewest uncertainties and, in the absence of further data, is the current proposed choice.

6.3.2.2 Advanced EVA Dust Handling Study

The lunar surface is covered with a fine dust-like soil to a depth of approximately 10 meters. Median soil particle size is 70 microns with 10 to 20 percent of the soil at 20 microns or less. The soil adheres to all surfaces of any exposed article. Due to the irregular shape and small size of the soil particles, they are extremely abrasive, scratching surfaces when attempts are made to remove them, and sand-blasting surfaces exposed to a ballistic stream of dust.

Lunar dust will present a hazard to EVA specifically by causing abrasion of exterior surfaces of extravehicular mobility units (EMUs), rovers, tools, and equipment, accelerating wear of bearings and seals, and scratching and fogging visors, lenses, mirrors, and sensor windows. Dust entrained during EVA and carried into the habitat will present similar problems to habitat equipment, including life-support systems. In order to effectively handle lunar dust contamination, an approach consisting of six elements must be used.

The first element, dust plume minimization, can be accomplished by designing equipment that operates in such a way as to prevent the formation of dust plumes and/or by adding dust shields to contain and confine any plumes that are produced. Dust shielding could be a small shield around a surface drill bit, or it might cover an entire worksite.

The second element is to design tools and equipment to be highly tolerant of lunar dust. They should first resist contamination and the accumulation of dust in sensitive areas via seal and seam design. Tools and equipment should also be easy to clean, and the proper equipment to clean them must be developed. Finally, abrasion-resistant materials should be used to minimize the effects of any dust that is picked up.

The third element is rover dust handling. Generally, it must be designed in accordance with the second element, described above, but it must also be specially designed to minimize dust plumes thrown up during transits of the lunar surface.

The fourth element is EMU dust handling, which is similar to the second element, specifically applied to the EMU. It also includes the important step of cleaning off the exterior of the EMU before airlock ingress to help minimize habitat interior contamination.

The fifth element is interior airlock atmosphere cleaning, which is performed to prevent contamination of the

habitat interior and is accomplished by cycling the air-lock atmosphere through filters and dust separators after repressurization but before hatch opening.

The final element is interior habitat and interior dust handling, which includes equipment design to resist contamination. Especially important is interior design to allow cleaning away accumulated dust.

6.3.2.3 Advanced EVA Suit versus Habitation Pressure Study

One goal for the lunar outpost is to maximize EVA productivity without compromising crew safety or health. EVA productivity is maximized by lowering EMU suit pressure as much as possible, down to a practical limit of about 3.8 psia. This adjustment also has the effect of reducing abrasions and other injuries to the crew caused by working against the pressurization-induced stiffness of the suit and suit gloves.

Unfortunately, going directly from a 14.7 psia habitat to a 3.8 psia EMU suit would cause the crew to experience decompression sickness including joint pains commonly called "the bends" and other, possibly fatal, dysbarisms. The cause of these difficulties is nitrogen absorbed into the body tissues at 14.7 psia, which evolves at lower pressures and forms bubbles in various parts of the body such as joints in the case of the bends and in blood vessels supplying the brain in the case of central nervous system dysbarisms. In the past, this problem has been avoided either by having a low habitat atmosphere (5 psia in Apollo/Skylab) or by pre-breathing a reduced cabin atmosphere for a time before each EVA in order to safely flush nitrogen out of the body (12 hour pre-breathe at 10.2 psia in the Shuttle program).

For productivity, it is highly desirable to eliminate any sort of pre-breathe. Although going directly from a 14.7 habitat to a 3.8 psia suit is not possible, other combinations of suit and habitat pressure can be used to enable a direct depressurization. A measure of the safety — and the limits — of any depressurization can be obtained by calculating the "R" value associated with it. R is defined as the partial pressure of nitrogen in the atmosphere (and the body tissues) before decompression, divided by the total atmospheric pressure after decompression. For each R value, an associated risk of bends can be determined and used as an approximation of total risk of all types of serious dysbarisms. For an R of 1.4, the incidence of bends is 5.75 percent per person per 8-hour EVA. For an R of 1.2, the incidence is 1.2 percent, and for an R of 1.0, the incidence is 0.08 percent.

With a habitat pressure of 14.7 psia, partial pressure of oxygen of 3.1 psia (normoxic condition), and an R value of 1.4, suit pressure would be 8.3 psia. This point is the upper limit of suit pressure for current technology. The 5.75 percent bends incidence for this case is too high for the large amounts of EVA envisioned with the lunar habitat. It would mean that a crew on a 6-month tour of duty could experience approximately 16 cases of the bends during that tour, some of them severe enough to require use of a hyperbaric chamber. The same difficulty exists with an R of 1.2, but it is reduced in magnitude to the point where the crew could expect only three cases of bends during their tour.

An R of 1.0 is preferred, since this level means that the crew can expect to complete their tour without experiencing the bends; however, the habitat pressure must be reduced. With a normoxic atmosphere, the highest cabin pressure compatible with an 8.3 psia suit and an R of 1.0 is 11.4 psia; however, it is highly desirable to lower the suit pressure as much as possible. If the suit pressure is lowered to 4.0 psia, the normoxic habitat pressure associated with an R of 1.0 is 7.1 psia.

NASA has established a 30 percent oxygen content (by volume) upper limit for habitat atmospheres because of flammability concerns. Above this limit, common materials suddenly become considerably more flammable. This point, for normoxic conditions, is exceeded with cabin atmospheres of 10.2 psia or less. This limit can safely be exceeded by using special nonflammable materials within the habitat. Skylab, for instance, operated safely for several months at a normoxic 5.0 psia.

The conclusion is that for safety and productivity, cabin atmospheres less than 14.7 psia should be considered along with suit pressures less than 8.3 psia. The ideal combination would probably closely resemble the 5.0/3.8 psia conditions of Apollo/Skylab.

6.3.2.4 Advanced EVA Life Support Impact Study

This section describes a study to evaluate the life-support equipment that would be necessary for EVA on an advanced mission. The areas investigated include venting/non-regenerable subsystems versus non-venting/regenerable subsystems in the portable life support system (PLSS), shorter duration PLSSs, and the impacts of additional life-support equipment on an emergency 2-day rover.

Non-venting/regenerable processes to provide life support are recommended for missions with numerous EVAs without resupply. The logistics costs of carrying resupply consumables for venting/non-regenerable processes can be prohibitive. For example, 500 to 2,090 kg of sublimator water would be required as resupply to support 280 EVA-astronaut days (6 months of regular EVA activity for two astronauts), depending on environmental conditions and astronaut size. This amount

is high compared to the resupply requirements for non-venting cooling systems, which would be in the range of 23 to 91 kg.

Currently, the baseline life-support equipment used for the Space Station Freedom PLSS includes non-venting and regenerable subsystems for CO_2 removal, power supply, and thermal control. The problem with this baseline PLSS is that it is estimated to weigh 195 kg; consequently, during a lunar EVA, the astronaut's muscles would need to support 325 Newtons. This number exceeds the OSHA limit of 200 Newtons for back-mounted equipment; the OSHA limit is currently being considered as the maximum PLSS weight requirement. One possible solution to the heavy PLSS problem would be to shorten the duration for which the life-support equipment must operate. Replaceable packs could be stored on the rover and changed out at appropriate intervals. Figure 6.3.2-1 shows the PLSS weight savings for shorter duration equipment using currently available technologies.

Technology improvements and alternative construction materials could help relieve the heavy PLSS problem. Other possible solutions include using a power-only umbilical and relocating some of the life support equipment to other parts of the suit.

The impact of a 2-day emergency capability on the rover has also been studied. This capability would allow two EVA astronauts to survive for 2 days before returning to the habitat. The PLSS and replaceable packs would supply life support for 8 hours, and an umbilical from the rover would provide the life support for the remainder of the 2 days. The rover umbilical would provide travel and emergency life support and could provide thermal recharge (refreezing the ice pack). The rover would use regenerable life-support equipment and would be recharged at the base. The replacement PLSS packs and the 2-day emergency capability will add approximately 900 kg to the rover.

In conclusion, the 2-day rover capability appears to be achievable with current technologies. Technology improvements or other solutions will be necessary to solve the heavy PLSS problem. Further study is recommended in the areas of PLSS weight requirements, the short duration pack, and PLSS materials of construction.

6.4 AUTOMATION AND ROBOTICS/HUMAN PERFORMANCE SPECIAL ASSESSMENT

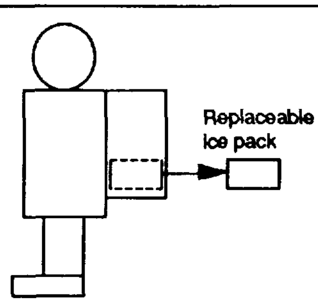
Special assessments in automation and robotics/human performance (A&R/HP) were directed at obtaining a better understanding of requirements in these areas for human lunar and Mars missions. In particular, analyses focused on barrier issues, critical problems, and high-leverage areas. Conventional and unconventional systems, technology, configurations, and technical options were identified and evaluated. The goal of the assessment was to provide system analysis and design capability to enable effective allocation of functions between humans and machines for human exploration missions. The study was an iterative process that incorporated new information as it became available.

To perform this assessment, workshops were held with each of the IAs to discuss issues and concerns. From this information, the barriers, critical technologies, and high leverage areas were identified and prioritized. In many cases, it was necessary to develop concepts for the critical opportunity areas to perform the high-level trade analyses, and these concepts provided a basis on which to evaluate the technology readiness level and to prioritize the technology needs. The high-level trade analyses addressed about 80 percent of the technology need items identified by the EXTWG.

The FY 1989 study efforts included engineering analyses of on-orbit operations and planetary surface systems. A crew size trade-off analysis and human performance evaluation (benchmarks) were conducted for representative operations at the orbital node, during spaceflight, and on the planetary surface. This section describes sev-

Weight of PLSS with downsized ice pack
(Power supplied by fuel cell)

Pack	Earth	Moon	Mars
8 hour	195 kg	33 kg	72 kg
4 hour	182 kg	31 kg	68 kg
2 hour	165 kg	28 kg	61 kg
1 hour	155 kg	26 kg	57 kg



Weight of PLSS with downsized ice pack, CO_2 removal unit, power unit
(Power supplied by battery)

Pack	Earth	Moon	Mars
8 hour	195 kg	33 kg	72 kg
4 hour	147 kg	25 kg	54 kg
2 hour	115 kg	20 kg	43 kg
1 hour	97 kg	16 kg	36 kg

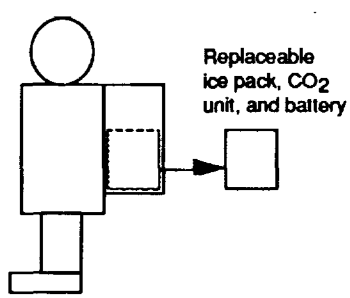


Figure 6.3.2-1.- PLSS weight savings for shorter-duration equipment.

eral concepts developed during FY 1989, including an evaluation of A&R technology areas defining technology readiness, priority, and critical applications.

6.4.1 Critical Concept Development

Critical concepts are capabilities that must exist to meet stated mission requirements. In general, these capabilities are enabling in that they overcome the limitations of EVA, reduce the number of crewmembers assigned to a particular task, and provide significant savings in crew time.

Figure 6.4.1-1 shows a software architecture to enable automated self-checking, monitoring, and maintenance of on-orbit and planetary surface systems. The user interface communicates with the user through a variety of media: natural language (both written and spoken), menu options, graphics, touchpanel, or any combination of these formats. The format of the information displayed to the user is matched to the detail required for users to effectively carry out tasks given varying levels of expertise. This flexibility enables a user to interact with the system from the highest command level to the lowest without having to be specifically trained in every aspect of the system. Although the user interface is predominantly procedural software for controlling graphics, menus, and keyboard handling, it also contains declarative software to process both verbal and keyboard-activated natural language requests. This software architecture would be applied to the critical concepts described below.

The automated rendezvous and "soft" docking of fuel tanks would use positioning sensors to coordinate movements and remotely manipulate the robotic arms. The robotic arms on the one vehicle then grasp the docking ring of the second vehicle. Crucial to this system is a real-time procedural software system that can process video data (for identification and guidance to docking target), identify hazardous conditions (such as EVA crew in docking path), and provide guidance, navigation, and control for docking operation with or without use of modular thruster packs. The system design is modular and the thruster packs, multi-arm units, and docking rings are interchangeable between tanks. The docking is considered "soft" by partial dissipation of the kinetic energy via conversion into potential energy by compression of springs.

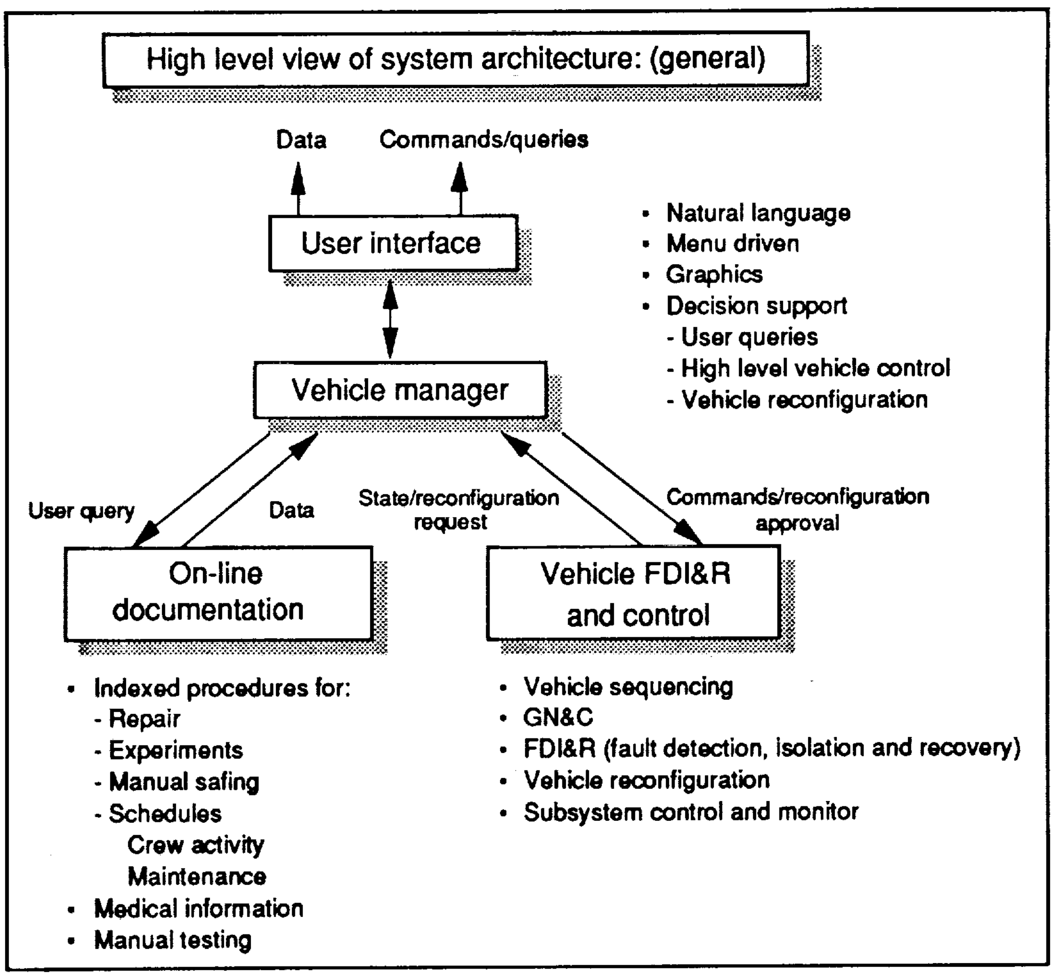


Figure 6.4.1-1.- Advanced software architecture.

Automated on-orbit refueling and propellant tank handling can use a helical concept, in which each propellant tank is automatically indexed and transferred from one carousel to the umbilical port for fuel transfer operations, and subsequently returned for storage. Within each propellant tank is a rotating mechanical device designed to generate internal pressure for fuel management. A feedback control system controls the angular velocity of the internal rotating mechanical device, based on sensor data of void development inside the propellant tank. Trend data analysis of void developments by expert system software is used to increase smooth propellant transfer. This system uses the automated soft docking concept described above and uses a fluid transfer line with ultrasonic sensors to control fuel transfer.

Due to the amount of construction, checking, maintenance, and repair activities associated with the exploration missions, a concept was developed for a versatile, modular, manned/unmanned multi-arm robotic vehicle. This concept has been called a space sub and is shown in figures 6.4.1-2, 6.4.1-3, and 6.4.1-4. The manned/unmanned space sub was designed to aid assembly of complex systems. It can be anchored to a truss or other structure, manipulate and position large or small structures, and be used for complex structure assembly. The system utilizes mature effector and sensor technology for control of the robotic limbs and can be equipped with

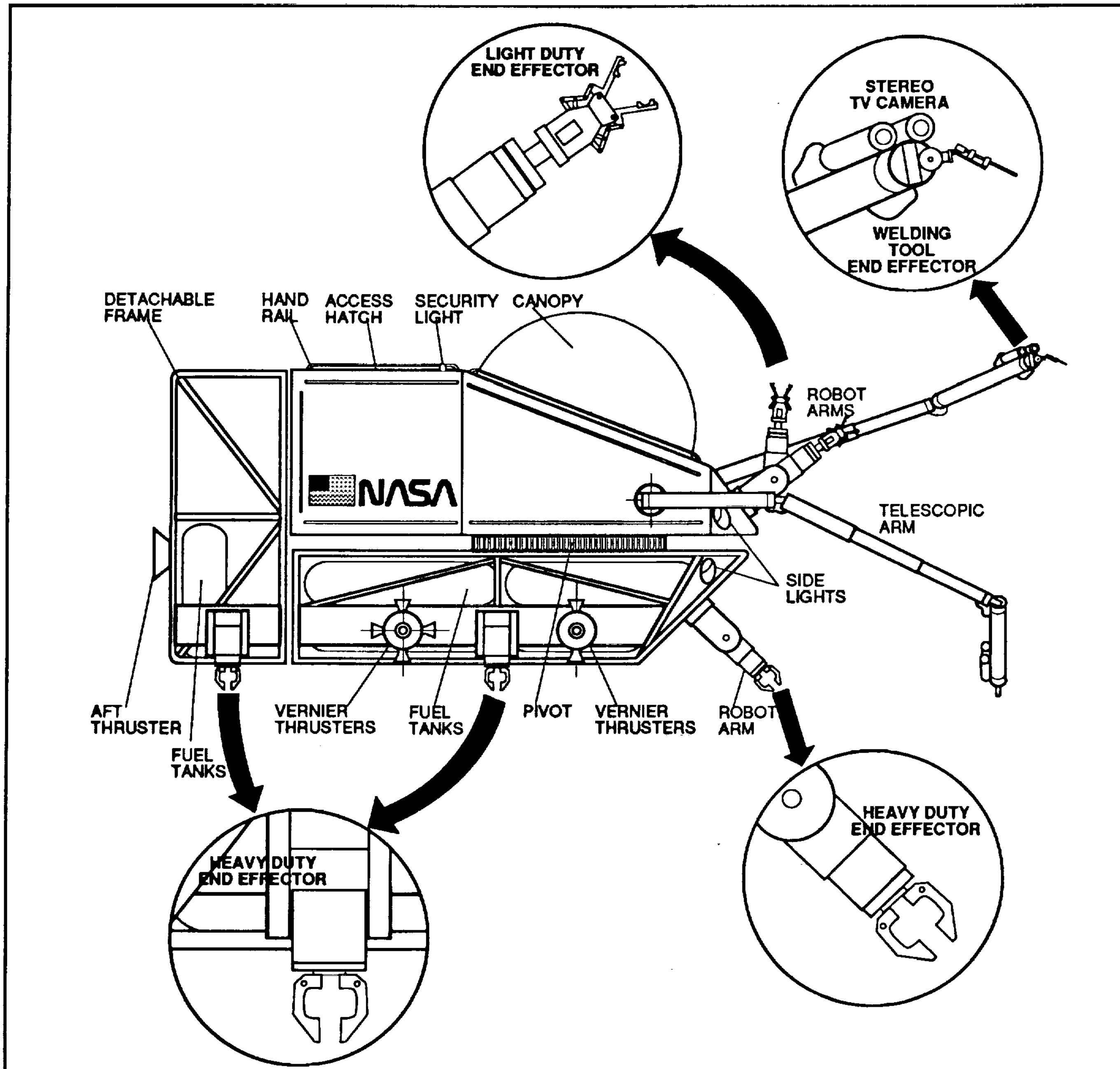


Figure 6.4.1-2.- Side view of space sub concept.

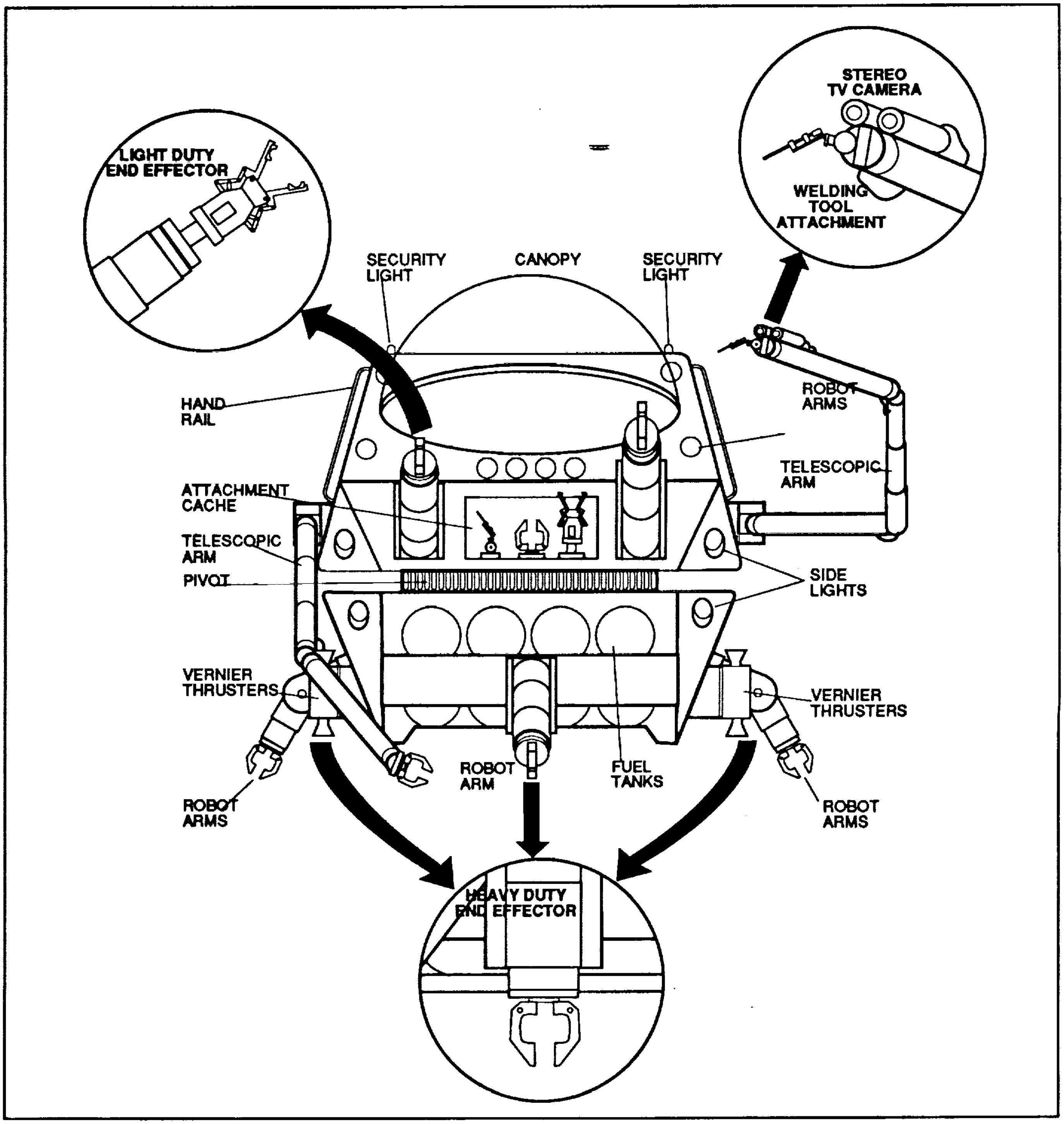


Figure 6.4.1-3.- Front view of space sub concept.

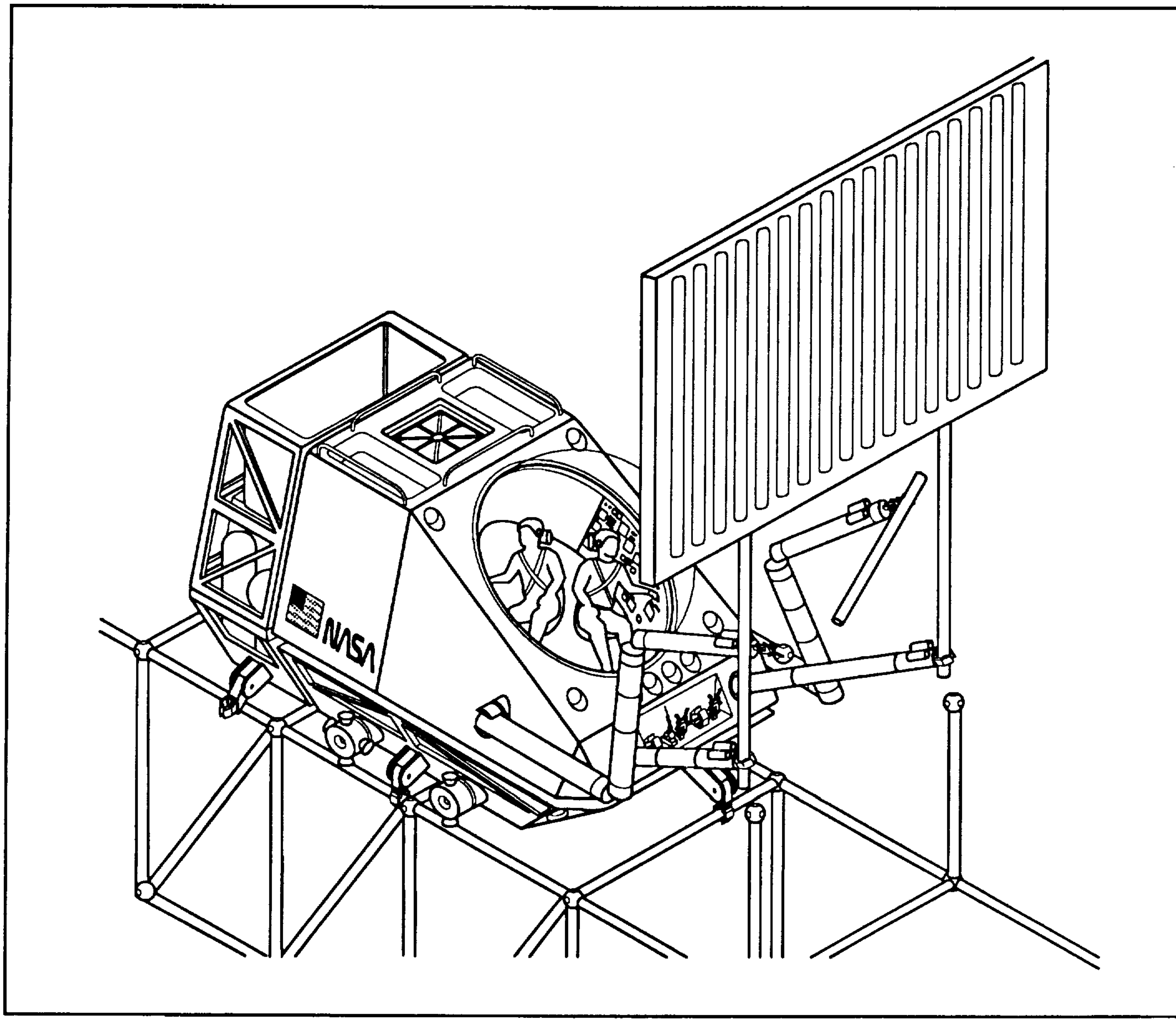


Figure 6.4.1-4.- Depiction of space sub assembly operations on truss structure.

an additional thruster pack for moving heavy loads. Expert system software is used to control the limbs to perform automated rendezvous and soft docking. Several video cameras are mounted on the space sub to aid in monitoring remote operations. Embedded fish-eye charge-coupled device sensors provide vision capabilities to end-effectors. The robotic vehicle, designed to be manned or unmanned, is modular (figure 6.4.1-5) and can be used as an in-space/ground transport power system for crew and equipment transfer, support of hazardous material handling, as a platform for special need equipment (e.g., science experiment deployment), and a variety of other applications. Expert systems are used to automate handling procedures, leaving human capability free for troubleshooting, inspection, and other critical requirements. On the surface, the space sub can be used for automated cargo unloading and deployment and for set-up of surface structures.

6.4.2 High-Leverage Concept Development

High-leverage concepts offer significant advantages, but are not critical for mission operations. Specific concepts in this category that have been derived for in-space and planetary surface A&R include the following:

a. Advanced engineering controllers for operation of space systems
b. Automated cargo unloading and deployment approaches
c. Automated landing site preparation
d. Automated habitat/radiation shield installation

e. Semiautonomous cooperating concepts for mining and LLOX production

f. Automated approaches for power handling and storage

g. Intelligent maintenance system

h. Automated dust contamination control

The concept was developed for an intelligent system to perform maintenance activities on space systems such as solar arrays or nuclear reactor units. The control system for this concept was designed with intelligent control, distributed problem solving, control strategies, and system-wide communications. The control system is made up of three subsystems, which control diagnostic activities, inspection functions, and robotic operations. The diagnostic subsystem uses model-based and case-based reasoning to perform diagnostic tasks. The inspection subsystem contains feature-based definitions, object recognition capabilities, and the ability to perform precision measurements. The robotic subsystem uses knowledge-based definitions and intelligent controllers to direct the "robot" to perform required maintenance tasks.

Figure 6.4.2-1 shows one of the concepts developed for automated mining. In this concept, the large triangular "straddler" (also used for unloading cargo from the lander) is utilized for slow excavation operations to grade, level, and expose flat undisturbed subgrade. It carries with it the miner/separator plant for mining the unprepared substrate and beneficiates during transport.

Figure 6.4.2-2 illustrates a lunar ilmenite oxygen reactor for the automated lunar LLOX production facility. The facility uses hydrogen reduction of ilmenite to remove oxygen from the regolith. Regolith is transported to the facility in a 27 t capacity hopper that travels on rails. Regolith is automatically transferred into the reaction vessel, where it is reacted with 900°C hydrogen gas. The hydrogen gas bonds with oxygen in the regolith, releasing water vapor, which is electrolyzed to recover the input hydrogen and obtain the oxygen product. This system has an estimated mass of 30 t and is landed intact on the surface.

Although these concepts are not essential to carry out the objectives of human exploration, they have potential to offer significant advantages over the current baseline designs. Further study is needed to assess the impacts of integrating these high-leverage concepts into mission scenarios.

6.4.3 Automation and Robotics Technology Areas

To define the A&R technology areas, several general barriers were identified.

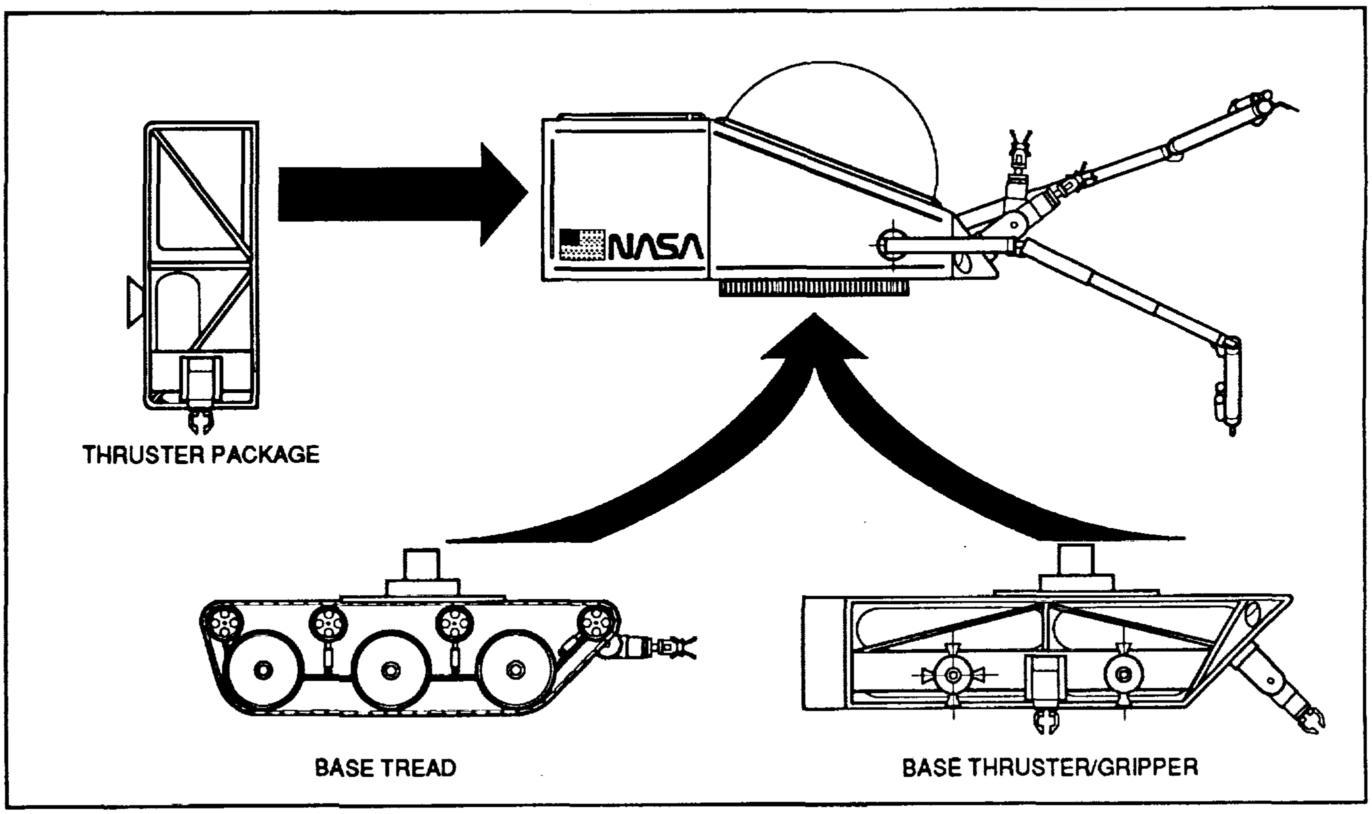


Figure 6.4.1-5.- Possible configurations of modular space sub.

a. Advanced A&R technology is limited for the increased complexity level of assembly, service, and repair.

b. No cost-effective approach exists for handling increasing reliance on A&R.

c. Analytical tools for optimizing mission operations are not available.

The A&R/Human Performance SAA began development of a computerized THURIS program, which is an analytical tool for optimizing mission design and operation. This program will be used to perform crew size trade-off analyses as a function of specific design and assumed level of A&R in a systematic fashion. Inputs and results of this application of the THURIS program are shown in figure 6.4.3-1. This program provides a starting point for future analytical tool development.

To better define other technologies to be considered for future development, table 6.4.3-I was constructed. This table includes technologies from the following A&R technology areas:

a. Supervisory systems

b. Sensing and perception systems

c. Actuator systems

d. Database systems

e. Executive control computer systems

f. Systems integration

For each of these technology areas, specific subsystems are defined, and the technology readiness and priority are assessed. Technology readiness values range from Level 1, where basic principles of the technology are observed and reported, to Level 7, where a system validation model of the technology has been demonstrated in space. In the need category, common and unique enabling technologies are labeled as I and II, whereas common and unique high-leverage technologies are labeled III and IV. In the timing category, a ranking of A is for near-term technologies; a ranking of C is for far-term. For the risk category, a ranking of 1 is for high-risk technologies, and a ranking of 3 is for low-risk tech-

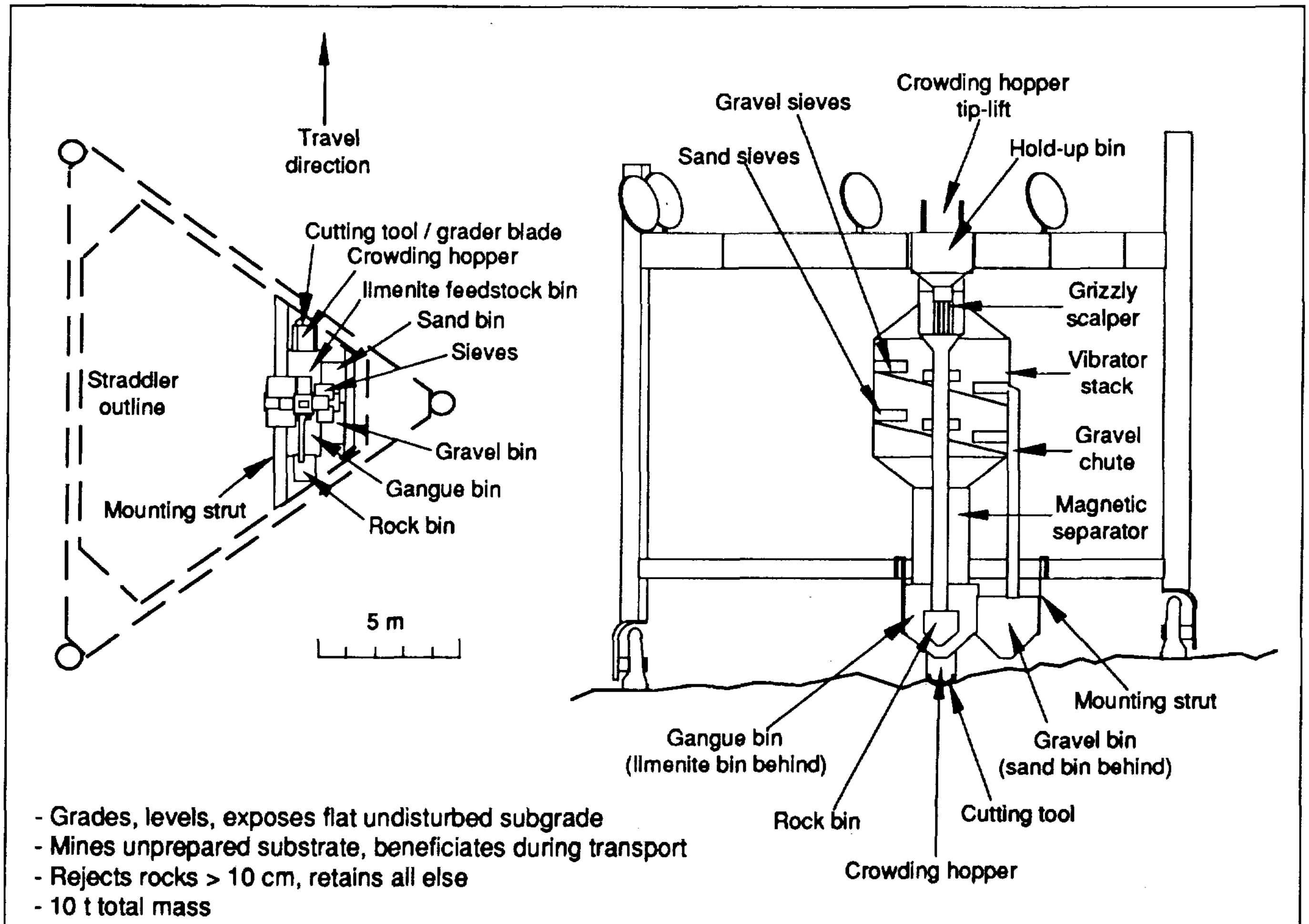


Figure 6.4.2-1.- Illustration of large straddler with miner/separator plant.

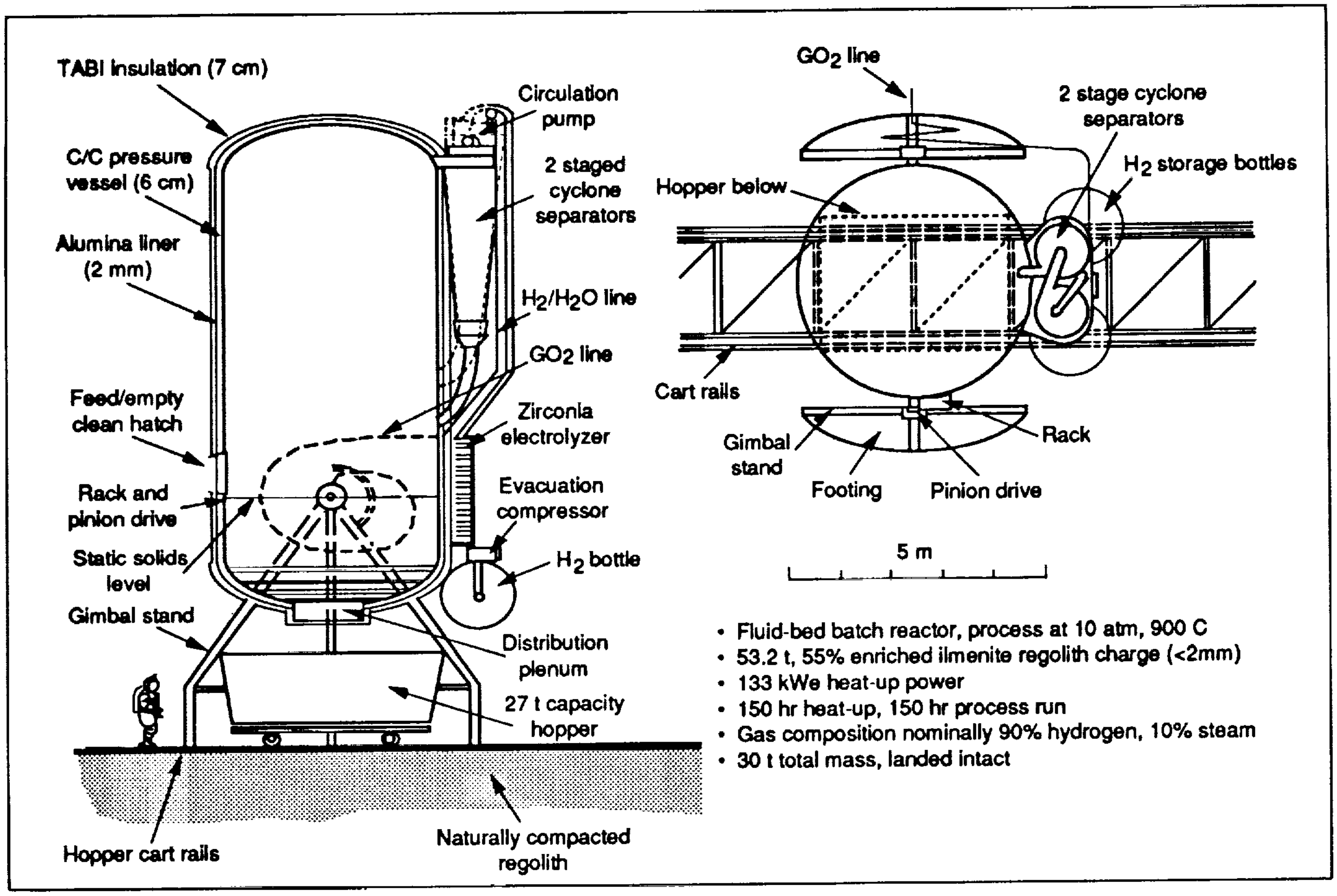


Figure 6.4.2-2.- Lunar ilmenite oxygen reactor.

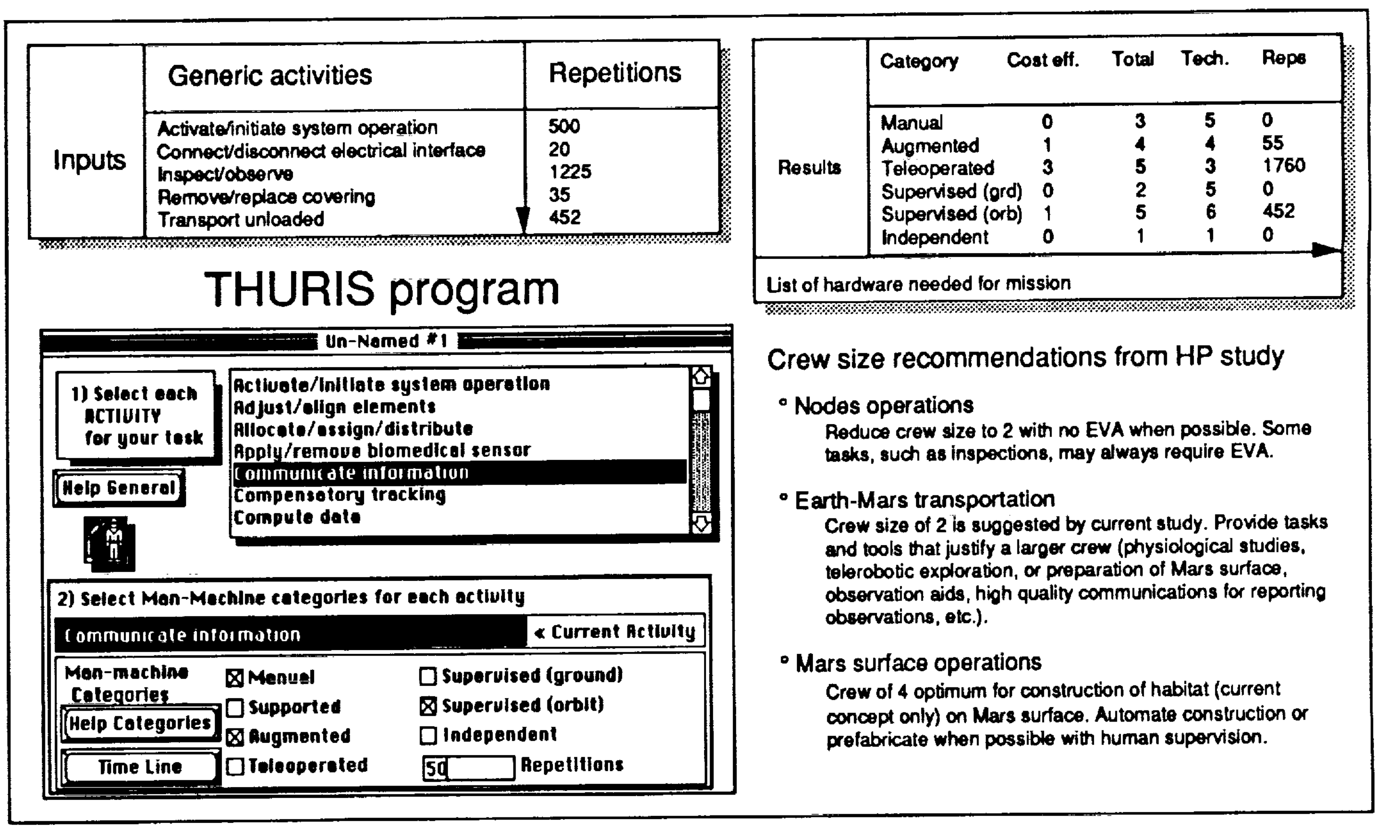


Figure 6.4.3-1.- THURIS program for crew-size tradeoff analyses.

nologies. This table can be used to determine requirements for development of various technologies that could be implemented in OEXP case studies.

As can be seen in table 6.4.3-I, the current A&R technology level cannot handle the increasingly complex mission scenarios. Moreover, the baselined A&R is not adequate for the construction, maintenance, and repair operations required for initial buildup and long-term operation of the space systems. Of particular concern is the effect of the environment (dust contamination) on long-term operation of the machines. What is required is a central focus of A&R for ensuring commonality and modularity between the on-orbit systems and the planetary surface systems. This central focus is necessary for achieving a practical, low-cost approach for providing the required A&R, and for ensuring long term maintainability of the space systems.

6.4.4 Summary

Table 6.4.4-I summarizes the activities of the A&R/Human Performance SAA during FY 1989. Included in this table are results/concepts presented to MASE and the IAs at scheduled working group meetings and program reviews. Also included are recommendations for future case studies and a summary of major conclusions drawn from study activities.

TABLE 6.4.3-I.- EVALUATION OF A&R TECHNOLOGY READINESS AND PRIORITY

A&R technology areas	Subsystems	Technology readiness		Priority level			Critical application
		Level	Dev. example	Need	Time	Risk	
Supervisory systems	Information feedback:						
	• Vision	7	Mission operations at: JSC, KSC, JPL	I	A	3	Remote S/C operations
	• Force	4	Teleoperator lab at: JPL, MSFC	III	B	2	Telemanipulations
	• Predictive display	3	MIT	III	B	2	Remote operations with short time delay
	• Modeling	2	ARC	III	C	1	Remote operations with long time delay
	Command input:						
	• Master replica	6	ORNL	III	C	3	Telemanipulation
	• Joystick	7	RMS	I	A	3	Telemanipulation
	• Digital	7	Space Shuttle	I	A	3	Remote operations
	• Voice	4	JPL, JSC	III	C	2	Multiple path control requirements
	• Integrated supervisory systems	6	JSC, KSC, JPL	I	A	3	Near-Earth and deep space mission operations
	• Human operators	6	JSC, KSC, JPL	I	A	3	
Sensing and perception systems	Sensors:						
	• Vision	7	Space Shuttle	I	A	3	Visual environment determination
	• Force/Torque	6	JPL, JSC	III	A	3	
	• Tactile	4	Case Western U.	III	A	3	Dexterous manipulators and other mechanisms
	• Slip	2	JPL	IV	C	3	
	• Range	6	JPL	III	A	3	3-D environment determination
	• Others	7	Space Shuttle	I	A		Special environments (vibration, deformation)
	Perceptors:						
	• Visual Image Identification and Recognition	1	U. of Michigan	III	B	2	Automatic pattern recognition and classification
	• Multisensor Identification and Recognition	1	U. of Michigan	III	B	1	Multisensor recognition and classification of information
	• Sensor fusion	1	U. of Michigan	III	B	1	
Actuator systems	Devices:						
	• Propusion jets	7	Previous S/C	I	A	3	
	• Switches	7	Previous S/C	I	A	3	
	• Electro motors	7	Previous S/C	I	A	3	MMV, OMV, OTV,
	• Vernier control	7	Previous S/C	I	A	3	Space Shuttle RMS, FTS,
	• Manipulators	7	RMS	I	A	3	Rovers, mobility units,
	• End effectors	6	RMS, JSC, JPL	I	A	3	Telerobotic mechanisms, etc.
	• Special purpose mechanisms	6	STEP, LaRC	I	A	3	
Database systems	Data:						
	• Integrated CAD/CAM	2	ARC	III	B	1	Database management system for mission operations
	• Technology management	3	GSFC	III	B	1	

TABLE 6.4.3-I.- (CONCLUDED)

A&R technology areas	Subsystems	Technology readiness		Priority level			Critical application
Database systems (continued)	Knowledge bases:						
	• Astronaut aids	1	JSC, ARC	III	B	1	Expert systems for operating and maintaining equipment in space
	• Operations monitor	6	JSC	III	B	3	Mission operations and control
	• Diagnostics	2	ARC, JPL	III	B	1	
	• Maintenance	1	ARC	III	B	1	Autonomous propellant production
	• Simulation	1	ARC	III	B		Surface rovers, free flyers,
	• World modeling	1	ARC	III	B	1	Construction equipment, etc.
	• Planning	2	ARC, JPL	III	B	1	
Executive control computer	Inference engines:						
	• Diagnostics	2	ARC, JPL	III	B	2	Autonomous propellant production
	• Planning	2	ARC, JPL	III	B	2	Surface rovers, free flyers
Systems integration	Systems:						
	• Coordinated control of multi-subsystems	2	ARC, JSC, LeRC	III	A	2	Automated self-check., monitor, maintenance, in-space vehicle processing, monitor and control of planetary surface station
	• Mobile A&R	2	JPL	III	A	2	Autonomous landing, rendezvous and docking, simple in-space assembly and construction, planetary science rover, application-specific A&R tools
	• Mobile cooperation A&R	1	ARC, JPL	III	B	3	Complex in-space assembly and construction, planetary surface operations (e.g., site preparation, mining, propellant production)

TABLE 6.4.4-I.- SUMMARY OF A&R/HUMAN PERFORMANCE SAA ACTIVITIES

Results/concepts used in FY 1989 case studies	Recommendations for future case studies	FY 1989 conclusions
Orbital node systems • In-space assembly of rigid aerobrakes is feasible. • Assembly and processing of interplanetary vehicles on-orbit are practical without excessive crew support. Transportation systems • Aerobrakes can be configured to utilize compressive forces during re-entry to keep the aerobrake system intact. • Lunar surface payloads are more readily accommodated with a lunar down-cargo capacity of 30 t (increased from 20 t) Planetary surface systems • Use of self-contained systems allows simpler emplacement operations. • Outpost buildup beginning with an unmanned, robotic cargo mission can allow emplacement of initial habitat and shielding. A large, three-legged, mobile gantry can be used for off-loading of the lander and for ground transport. • Surface vehicles can be operated using low power (~ 5kW). • Using separate regolith-shield containers is more desirable than direct module burial for radiation protection. • Solar arrays for surface power can be robotically deployed using rigid-plate, tripod-mounted solar array units.	Orbital node systems • Development of a versatile manned/unmanned multi-arm robotic vehicle (space sub) is needed for construction, checkout, servicing, repair, and rescue operations. This system can also be outfitted for use on planetary surfaces. Transportation systems • ETO vehicle payload shroud should be increased from 12 m to 15 m to accommodate large, automatically deployed payloads. Planetary surface systems • LLOX production should begin early using solar power sources. • LLOX reactor should be designed for ease of robotic control. • Mining power requirements may be reduced by excavating slowly with a shallow scraper. • Failure modes and effects analysis should be conducted for surface system hardware.	• All recurring operations and definable tasks, such as systems monitoring, fault diagnosis, house-keeping, etc., should be automated to the fullest degree possible. • Unique operations and tasks requiring circumspection, i.e., artificial intelligence, should generally not be automated, unless automation is the only means to provide the capability. • A&R/human performance considerations must be included from the start of conceptual design. • Commonality and modularity should be used wherever possible in both hardware and software. • A general purpose multi-arm robotic vehicle (space sub) is necessary for evolutionary type exploration missions. • Precursor missions and systems, such as Space Station Freedom, MMV, FTS, OMV, HLLV, Mars Observer, Lunar Observer, Mars Rover/Sample Return, etc., will provide a broad technological base for exploration type missions. • Outstanding areas relating to OEXP case study requirements that require additional engineering development include: (1) automated propellant management and refueling in space; (2) automated construction or assembly of large aeroshells in space; (3) automated assembly and construction on the Moon and Mars including loading and deployment; (4) automated mining and propellant production.
Sources: - A&R/Human Performance Annual Report, Volume VI-Special Studies - Video/Scripts - "Getting There" and "Manned/Unmanned Space Sub" - Engineering Analysis for Assembly and Checkout of Space Transportation Vehicles in Orbit - Engineering Analysis for the Design, Emplacement, Checkout and Performance of Robotic Lunar Surface Systems - Reliability Evaluation of a Robotically Constructed Lunar Base		- FTS Capabilities Assessment for On-Orbit Operations and Planetary Surface Operations - Advanced Engineering Software for In-Space Assembly and the Manned Mars Spacecraft - Human Performance Crew Size Study - Computerized A&R/Human Performance Tradeoff Tool (THURIS) - Structural Integrity/Smart Sensor Assessment - Augmented Pressurized Rover - Design Guidelines and Rules

6.5 EARTH-MOON NODE LOCATION

Most lunar outpost studies to date have presupposed the use of a transportation system similar to the one used for Apollo: a separate Earth-Moon transportation vehicle and lunar lander with payload transfer occurring in low-lunar orbit (LLO). This strategy is used because it can provide the best mass performance for the payload delivered to the lunar surface. In an operational scenario in which vehicles will be reusable and cost becomes a dominant factor, a different strategy may be more appropriate. The objectives of the Earth-Moon node location trade study are to determine if such an alternative strategy exists and to characterize it in terms of performance, functionality, and cost. This objective can be broken into five main components, which are discussed below.

The first component is to determine at what activity/capability level in the development of a lunar outpost an Earth-Moon libration point or lunar orbit node will contribute to program cost avoidance. "Node" in this context could consist of as little as a rendezvous point or as much as extensive permanent facilities. Such facilities could provide payload warehousing, propellant storage and transfer, on-orbit maintenance, and/or vehicle hangars.

The second component is to determine the best location in Earth-Moon space for this node. Figure 6.5-1 illustrates options for this location. Both mass performance and cost will be determining factors in this selection.

The third component is to determine the capabilities a node must have to support a lunar outpost. In other words, where in the spectrum of node facility capabilities discussed above is the best combination to support a permanent lunar outpost?

In any operational scenario, Space Station Freedom will serve as a transfer point between the Earth-to-orbit and the Earth-orbit-to-lunar-surface transportation systems. As such, Space Station Freedom will initially have whatever permanent on-orbit facilities are needed to support the space-based segment of the transportation system. The fourth component of this study is to determine at what flight rate and/or activity level it becomes advantageous to move or duplicate these facilities at an orbital location other than Space Station Freedom.

The last component of this study is to determine the best location in the Earth-Moon system for a node to support the assembly, propellant resupply, and launch of a piloted mission to Mars, assuming the availability of lunar-derived propellants.

6.5.1 Background

Portions of the issues discussed above were examined last year as they applied to the four case studies then under consideration by the Office of Exploration. Results are documented in Volume II of the Exploration Studies Technical Report, FY 1988 Status (Technical Memorandum 4075). Significant points relevant to this year's study are summarized below.

The requirement for and function of an orbiting node were assessed for all FY 1988 case studies. In each case, the evaluation criteria were mass performance and/or operationally enabling functions; cost was not evaluated. For the first two FY 1988 case studies (Phobos and Mars expeditions), no node beyond low-Earth orbit (LEO) was deemed necessary. For the Lunar Observatory case study, the choice was defined from the outset as either a LEO node or a LLO node. The conclusion reached was that a low-Earth orbit node is preferable. The fourth FY 1988 case study (Lunar Outpost to Early Mars Evolution) was not examined in sufficient depth to determine whether a node other than one in low-Earth orbit would be required.

The location of an Earth-Moon node is a question that has been considered since the late 1960s. In 1969, the President's Space Task Group studied the necessity for and characteristics of a lunar space station as part of its integrated program plan for exploration of the Moon in

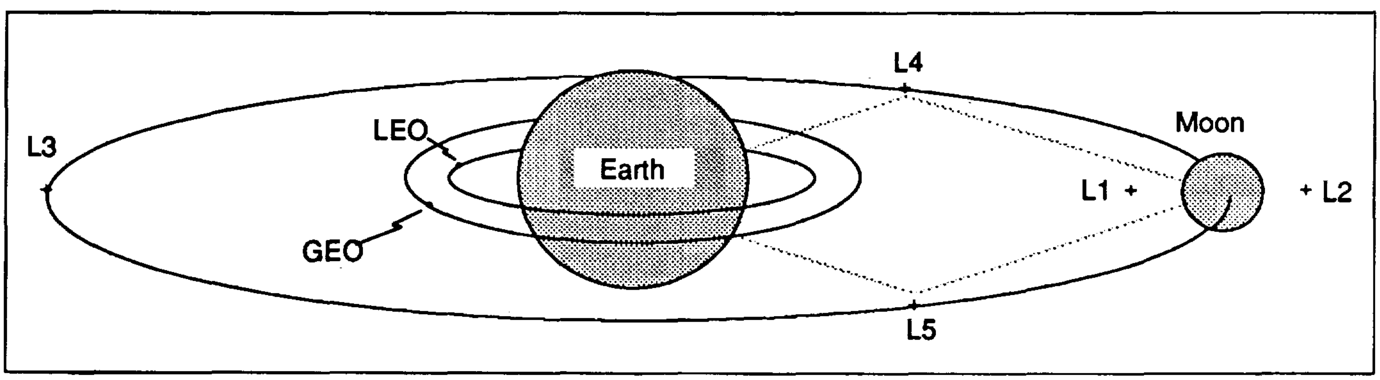


Figure 6.5-1.- Potential node locations in Earth-Moon space.

C-3

the 1980s and beyond. This plan recommended placing such a space station in a 60 nautical mile altitude polar orbit to allow global access to the Moon's surface. Proposals were also made to locate this facility at the L1 or L2 Earth-Moon libration point.

A study conducted by A.D. Little in 1987 examined the economics of a lunar transportation system, including the best node location. The system was designed primarily for delivering lunar-produced material (principally lunar-derived liquid oxygen) to LEO. Conclusions reached by this study were heavily dependent on the assumptions for the Earth-to-orbit launch costs and their effect on the return on investment for such a transportation system. These assumptions drove the study to use electromagnetic and/or laser-assisted propulsion to achieve the highest efficiencies possible. As a consequence, the results did not provide a high degree of insight into this problem, but provided useful economic considerations and analytical processes.

Several other studies have also indicated that it is not profitable from a performance standpoint to return lunar material (including lunar-derived liquid oxygen) to LEO. Returning material, even lunar-derived liquid oxygen, increases the size of the structure and aerobrake shield that must first be pushed to the vicinity of the Moon. The penalties associated with returning material outweigh the benefits derived. There are advantages, however, to using lunar-derived liquid oxygen for propulsive maneuvers near the Moon. Using lunar-derived liquid oxygen alleviates the need to bring this propellant through the logistics train from Earth.

These previous investigations have helped this study by indicating the activities that may be carried out at a node, the cost factors that should be considered, and the scenario(s) that is likely to be the most cost-effective.

6.5.2 Study Approach

Initially, this study assumed that a minimum transportation infrastructure will provide the lowest development and operational cost to support a lunar outpost. Space Station Freedom was assumed to be part of this minimum infrastructure. Study tasks were then constructed to evaluate the need for multiple vehicles and a cislunar node to support a lunar outpost. From these studies the "best" (i.e., lowest cost or operationally enabling) transportation scenario and node location would be identified. Specifically, the tasks include:

a. Determine the rank order of Earth-Moon staging locations for support of a lunar outpost using mass performance as the criterion.

b. Determine the operational activities (if any) that warrant facilities at a cislunar node.

c. Determine at what flight rate or activity level a cislunar node contributes to cost avoidance.

d. Prepare conceptual designs for cislunar node facilities to support a lunar outpost.

e. Assuming lunar-derived liquid oxygen production, determine the rank order of node locations in Earth-Moon space to support a piloted Mars mission.

The trade space for these tasks consists of six transportation scenarios and six cislunar node locations. Table 6.5.2-I defines these options. Within each scenario, all vehicles are assumed to be reusable and to use aerobraking where feasible. The use of lunar-derived liquid

TABLE 6.5.2-I.- TRANSPORTATION SCENARIO AND CISLUNAR NODE LOCATION DEFINITION

Transportation scenario

1. Direct from low-Earth orbit to the lunar surface; no lunar-derived liquid oxygen for propellant.
2. Direct from low-Earth orbit to the lunar surface; lunar-derived liquid oxygen is available for Earth-return propellant.
3. Separate space transfer vehicle and lunar lander; lander based in low-Earth orbit; no lunar-derived liquid oxygen for propellant.
4. Separate space transfer vehicle and lunar lander; lander based on lunar surface; no lunar-derived liquid oxygen for propellant.
5. Separate space transfer vehicle and lunar lander; lander based on lunar surface; lunar-derived liquid oxygen is available to lander as propellant.
6. Separate space transfer vehicle and lunar lander; lander based on lunar surface; lunar-derived liquid oxygen is available to lander and to space transfer vehicle (for Earth return only) as propellants.

Node location

1. None
2. Geosynchronous Earth orbit
3. Libration point #1 (L1)
4. Libration point #2 (L2)
5. Libration point #3 (L3)
6. Libration point #5 (L5)

oxygen is one of the options that differentiate the scenarios. Scaling factors for the vehicles are the same as those used for the propellant leveraging study described in the next section. Life-cycle cost (i.e., development plus operations costs) is used as the discriminator for the 22 possible combinations of node location and transportation scenario.

6.5.3 Study Results

For the first task, several sources were reviewed to compile a list of delta V values between node locations under consideration. Small variations in the values were common, depending on the assumptions made by those conducting the study. The A.D. Little study developed a complete set of delta V values generated with a consistent set of assumptions. These values are listed in table 6.5.3-I, and they will be used in subsequent tasks.

A set of linear equations was then constructed that describes each of the six transportation scenarios. These equations were constructed in a manner such that vehicle performance factors could be varied as part of a parametric sensitivity study. Once in this form, the equations can be solved simultaneously.

Table 6.5.3-II lists the scaling factors used as "baseline" values. When combined with the delta V data from table 6.5.3-I, the resulting mass performance for each of the 22 combinations is shown in figure 6.5.3-1. These data have been non-dimensionalized by dividing the transportation system mass (vehicle plus propellant) by the payload delivered to the lunar surface. This ratio characterizes the overhead required to deliver each kilogram

TABLE 6.5.3-I.- DELTA V BETWEEN VARIOUS NODE LOCATIONS

From \ To	LEO	GEO	L1	L2	LLO	Lunar surface	L5	Escape
LEO		4.33	3.77	3.43	4.04	5.93	3.97	3.22
GEO	2.06		1.38	1.47	2.05	3.92	1.71	1.30
L1	0.77	1.38		0.14	0.64	2.52	0.33	0.14
L2	0.28	1.47	0.14		0.65	2.53	0.34	0.71
LLO	1.31	2.05	0.64	0.65		1.87	0.98	1.40
Lunar surface	2.74	3.92	2.52	2.53	1.87		2.58	2.80
L5	0.84	1.71	0.33	0.34	0.98	2.58		0.43

Note: All figures in km/s.

(From "ASTS Study," A.D. Little, Inc., ADL Reference 54104, Dec. 1987)

TABLE 6.5.3-II.- BASELINE SCALING FACTORS FOR TRANSPORTATION SYSTEM ELEMENTS

Scaling factor	Direct-to-surface STV/lander	STV	Intermediate node lander
Mix ratio	5.16	5.16	11.00
Isp	482.00	482.00	409.18
f1, f3	0.11	0.11	0.11
fs, f2	0.12	0.10	0.02

Key:
- Mix ratio: Oxidizer/fuel mixture ratio
- Isp: Specific impulse (sec)
- f1, f3: Tankage structural factor (kg tank/kg propellant)
- fs, f2: Shield (fs) and lander (f2) structural factor (kg/kg)

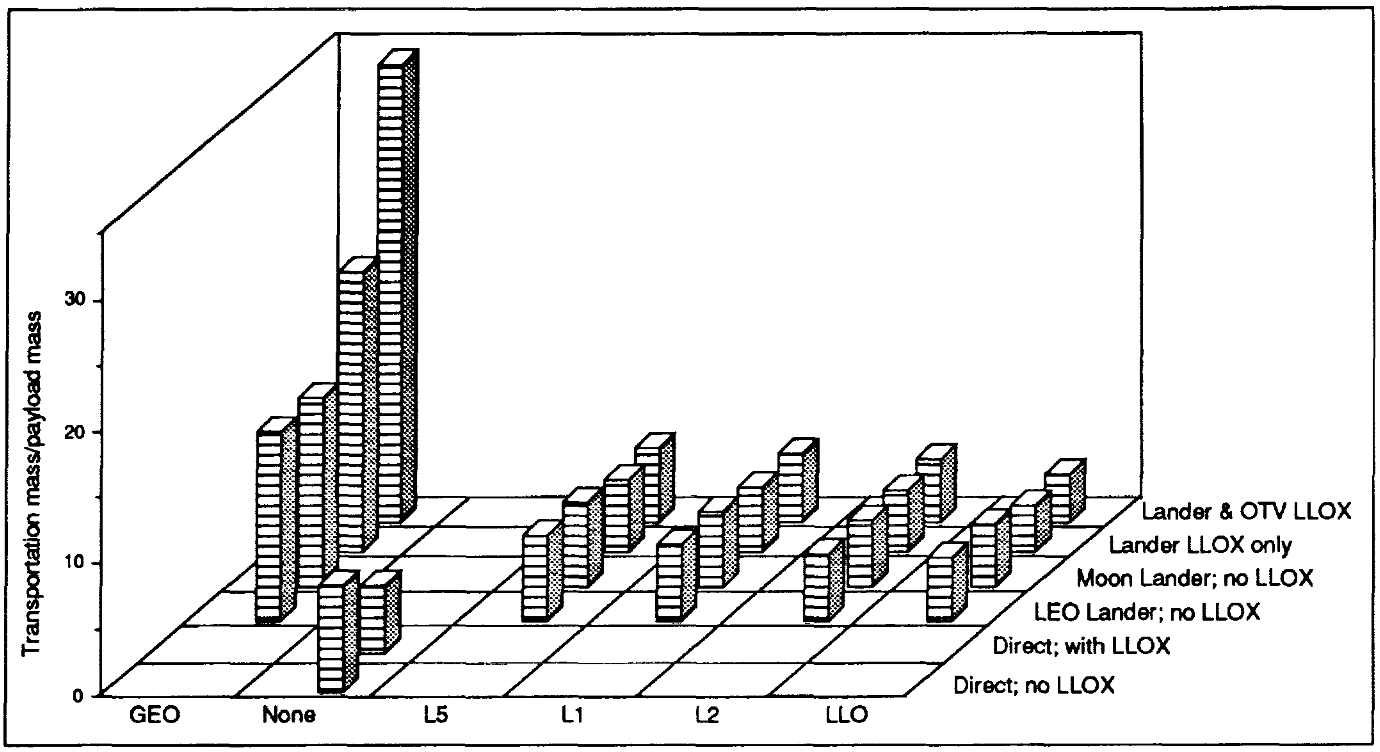


Figure 6.5.3-1.- Transportation system mass performance for all options.

of payload to the lunar surface. Minimizing this overhead provides a first-order indication of the most cost-effective node location. As can be seen in figure 6.5.3-1, the GEO node location requires the highest overhead of the options considered. Figure 6.5.3-2 illustrates the same data without the GEO node. Based on these results, the three best locations, in ranked order, are LLO, L2, and L1. For each of these options, it can be seen that the use of lunar-derived liquid oxygen as propellant for a lander based on the surface produces the best results. The use of lunar-derived liquid oxygen for return of the orbit transfer vehicle to low-Earth orbit actually increases the overhead, because the cost of transporting hydrogen to the Moon to lift the additional oxygen from the Moon exceeds the savings derived from using oxygen for the return flight to low-Earth orbit.

For the second task, previous studies were reviewed, and new information was gathered from those working on the lunar outpost case study. The results can be divided into five categories, which are discussed below.

The first category consists of those operational advantages that are obtained from having no node other than LEO. In this case, all on-orbit maintenance for a lunar lander and/or space transfer vehicle can be performed at a common set of facilities; no duplication is needed in another space-based facility or on the lunar surface. Operations are simplified, since an entire vehicle stack can be assembled, fueled, and checked in one location. All vehicles then return to the same facility for disassembly and repair (if necessary).

In the second category are those functions common to all nodes beyond low-Earth orbit. Facilities at this node can provide cargo handling and temporary payload storage, relieving the transportation system from carrying hardware to accomplish these functions. Temporary storage also opens the opportunity for optimizing the payload size and flight rate for the transportation system elements, since these vehicles would not be obligated to meet other elements at specific times. A communication link between various lunar surface sites or between the lunar surface and Earth could be provided at a node. Finally, if pressurized facilities are available, the node can function as a rescue point and safe haven.

The next two categories apply to low-lunar orbits. If the facility is located in a polar orbit, access to the entire lunar surface is possible. Near-polar latitudes can also be accessed once per orbit, whereas near-equatorial sites can be accessed on a 14-day cycle. Such an orbiting facility could be used for remote sensing of the surface and data gathering from surface sites.

For a facility in a near-equatorial orbit, access can be attained on every orbit. Such an orbit would allow large electromechanical launchers to be used, improving the surface-to-orbit efficiency.

The last category applies specifically to libration point nodes. Delta-Vs to and from interplanetary trajectories

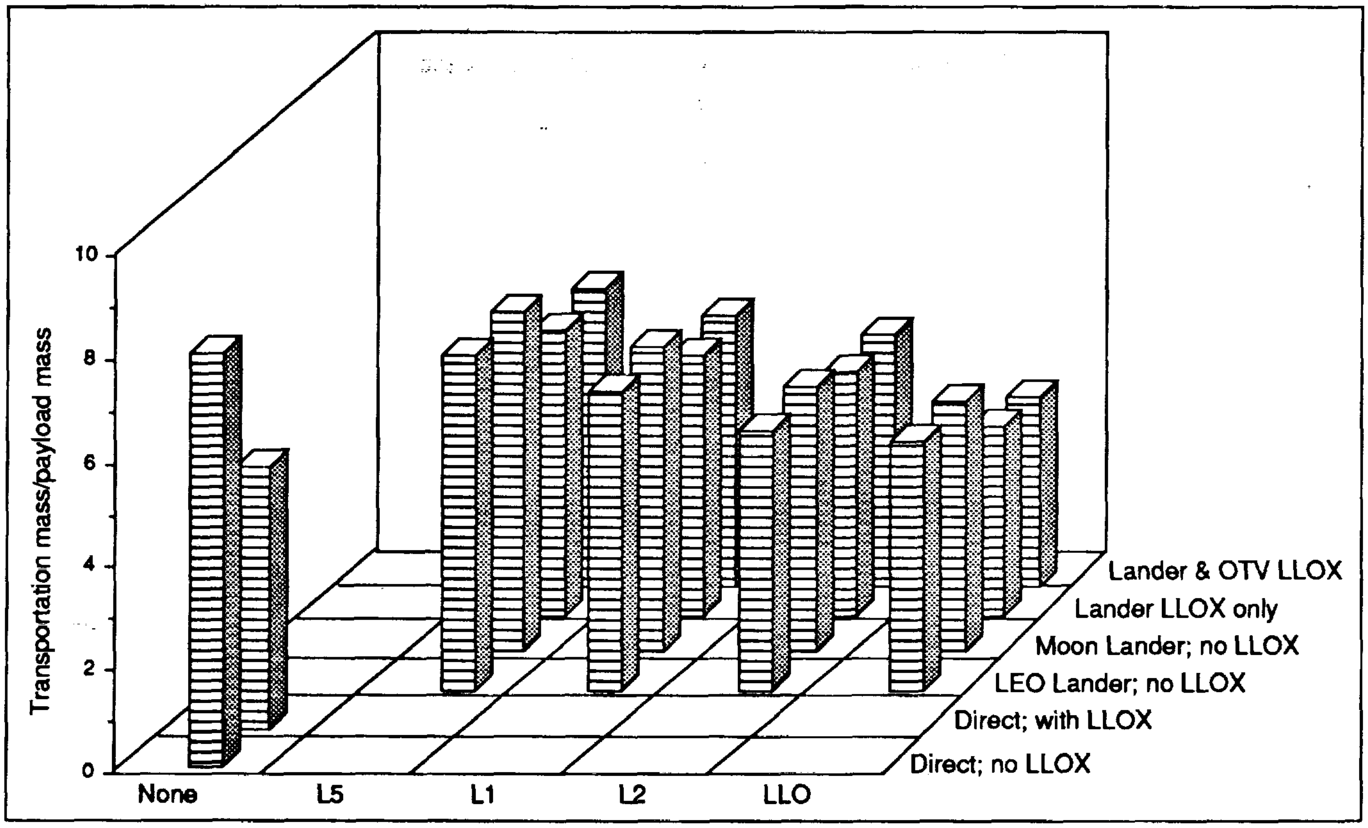


Figure 6.5.3-2.- Transportation system mass performance for selected options.

are typically the lowest from these points when compared to other node locations. The L2 point combines the access advantages of the low-lunar orbits discussed above, providing both global and continuous access to the surface. The drawback in this case is the flight time required between the surface and this point, typically 3 days. Electromechanical launchers can also be used to send payloads to the L5 point.

For the third task, life-cycle cost was used as the criterion to measure which node, using which scenario, and at what flight rate provided the most cost effective option to support the lunar outpost. Costs used to evaluate these combinations were assumed to include three components: (1) development and production costs for hardware items including transportation systems, surface systems, and any node facilities; (2) transportation cost from Earth's surface to the lunar surface; and (3) operations costs for the space-based transportation system and facilities used to support this transportation system, both in space and on the lunar surface. To carry out this analysis, the approach used was to first examine the effect of significant cost drivers, specifically Earth-to-orbit launch costs and the cost of operating lunar surface facilities (e.g., liquid oxygen production plant). Once the most promising options have been identified, other costs can be added to determine if the trends remain the same.

In researching this task, it became obvious that cost estimating relationships for lunar surface equipment or operation of space-based systems do not exist. A study reported in section 6.4 investigated costs associated with these activities, but the results were not available in time to be used here. For purposes of this study, conservative estimates were made using known data. Specifically, all hardware was estimated to cost \$50,000/kg (comparable to the development cost for the Hubble Space Telescope), and Earth-to-orbit costs were estimated to be \$10,000/kg (based on \$300M/launch for the Space Shuttle with a 27.2-t payload on board).

For those cases in which no lunar liquid oxygen was used, only the Earth-to-orbit costs were tracked. When lunar liquid oxygen was used, the additional cost associated with setting up and maintaining the production plant was also tracked. Items costed for this plant include fixed facilities (consisting of the production plant, power plant, and a prorated portion of the habitat for support personnel) and operating supplies (consisting of spare parts for the plant and a prorated portion of the crew consumables). Table 6.5.3-III lists the assumed scaling factors that are proportional to the annual output of the liquid oxygen production plant.

To gauge the effect of these costs on the performance of the transportation scenarios and node locations, a steady-

TABLE 6.5.3-III.- SCALING FACTORS CHARACTERIZING A LUNAR LIQUID OXYGEN PRODUCTION PLANT AND SUPPORT FACILITIES

Fixed facility elements	
Liquid oxygen production plant	1/5 plant mass/LLOX output/year
Power plant	1/10 plant mass/LLOX output/year
Habitat facility	1/10 habitat mass/LLOX output/year
Combined	4/10 kg total facility/kg LLOX output/year
Resupply/maintenance elements	
Spare parts	5% of plant mass/year + 2.5% power
	Plant/year + 2.5% habitat/year
Consumables	2,700 kg/person/year; 2 people/100,000 kg LLOX/year
Combined	7/100 kg resupply/kg LLOX produced/year

state delivery of payload to the lunar surface was modelled for 10-year and 20-year periods. Performance values discussed earlier were used to characterize the efficiency of each transportation scenario. The results from this analysis are shown in figure 6.5.3-3.

The first effect to be noticed is that those cases using lunar-derived liquid oxygen consistently require lower overall cost when compared to those cases that do not use this resource. However, the crossover point, defined as the time at which the accumulated cost of one option becomes greater than the other option in the comparison, can occur in as few as 2 years or as many as 15 years. A more interesting point to note here is that the direct-to-surface case using lunar-derived liquid oxygen is reasonably competitive with the overall best case: a low-lunar orbit node with a surface-based lander using lunar-derived liquid oxygen. This direct-to-surface case removes the need to stage the transportation vehicle at some intermediate point, simplifying operations. This option also allows the entire vehicle plus payload to be checked prior to launch, and it allows the transportation vehicle to be maintained at a single facility based in low-Earth orbit. These considerations could make the additional operating costs worthwhile. These results are promising and should warrant further investigation.

Analyses for the fourth and fifth tasks were not carried out during this year. Activity in these areas may continue in the future. Based on the analyses that were accomplished, several conclusions and recommenda-

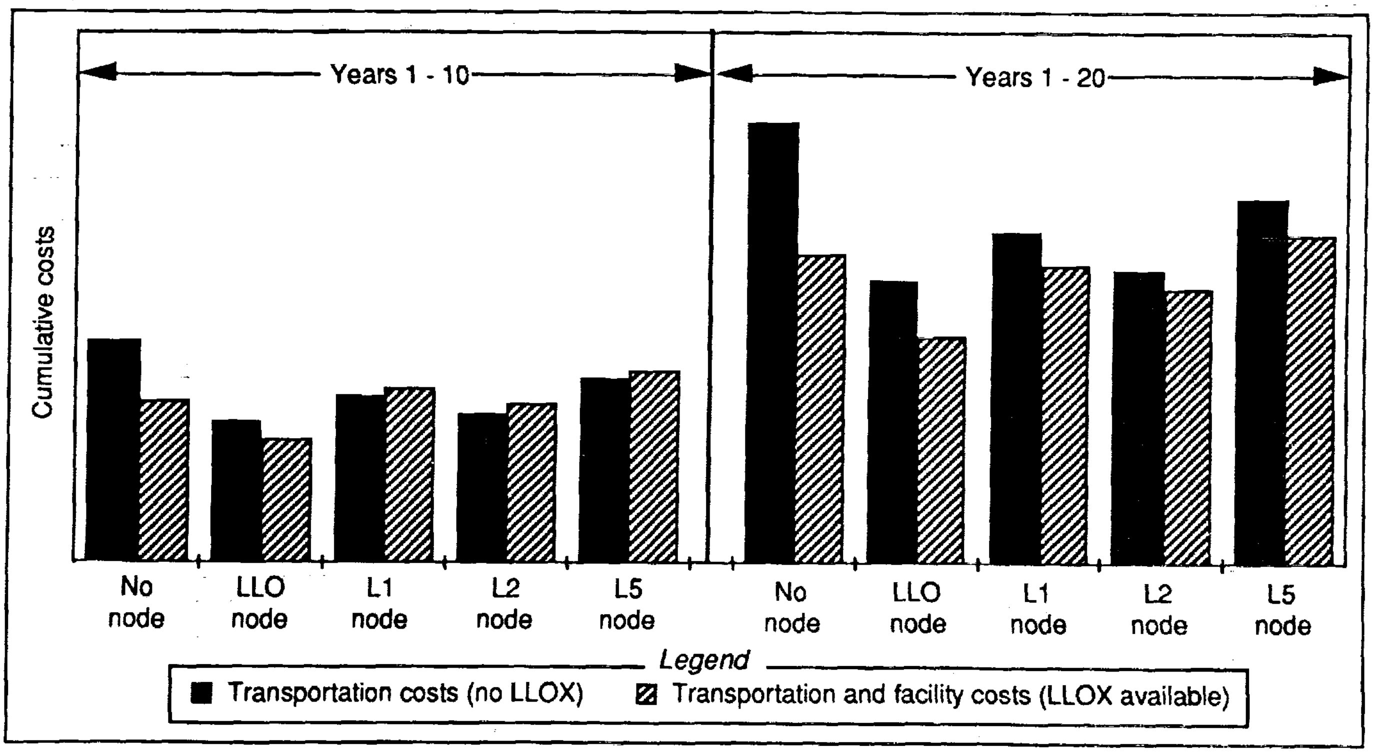


Figure 6.5.3-3.- Accumulated annual costs for the direct-to-surface scenario and selected node locations using the lunar surface-based lander transportation system.

tions can be made. First, based strictly on mass performance, the three best node locations, in ranked order, are low-lunar orbit, L2, and L1. The performance for each of these locations can be noticeably improved with the use of lunar-derived liquid oxygen to assist parts of the transportation system. However, the mass performance of these locations is sufficiently close that life cycle costs could alter the conclusion about which is best. In fact, a simplified analysis has indicated that the L2 point may be more cost-effective to operate than the low-lunar orbit location. Introduction of lunar-derived liquid oxygen also makes a direct-to-lunar-surface option competitive on a mass performance basis and brings with it the advantage of concentrating space-based facilities in low-Earth orbit, introducing more potential cost savings. Additional work is required to adequately characterize the hardware and operations costs for the elements in this system before a definitive answer as to the best location for an Earth-Moon location can be provided.

6.6 LUNAR LIQUID OXYGEN LEVERAGE

The objective of this trade study was to determine the lunar outpost activity level at which the production of oxygen from lunar materials becomes attractive as a means of reducing total program costs. Presumably, some level of demand for oxygen exists wherein a return on investment is possible by producing that oxygen on the Moon rather than importing it from Earth, as shown in figure 6.6-1. Past studies have presented different conclusions about where this oxygen demand level exists, if at all, primarily because of the different assumptions made in the various studies. The question of when a return on investment is possible when producing lunar oxygen raises a complex series of related questions: What are the objectives of the lunar outpost? Where is the oxygen to be used — low-Earth orbit (LEO), low-lunar orbit (LLO), or on the lunar surface? How will the oxygen be extracted from the lunar regolith? Are there any other usable by-products such as lunar-derived fuels, metals, or volatiles? What is the overhead associated with producing the oxygen; i.e., how many additional crew are required on the surface, how much additional power is needed, and how much additional equipment must be taken to the Moon every year? What is the transportation architecture between Earth and the Moon — all-chemical, nuclear, mass drivers on the Moon? What are the development, production, operational, and transportation costs for all the required additional systems? Each of the studies conducted to date has examined some of the above questions. Rarely, however, have any of the studies looked at the same set of questions, thereby making impossible direct comparisons of results. In addition, few studies have used cost as the principal discriminator of results; most have used mass to LEO as a first-order cost estimate, in order to

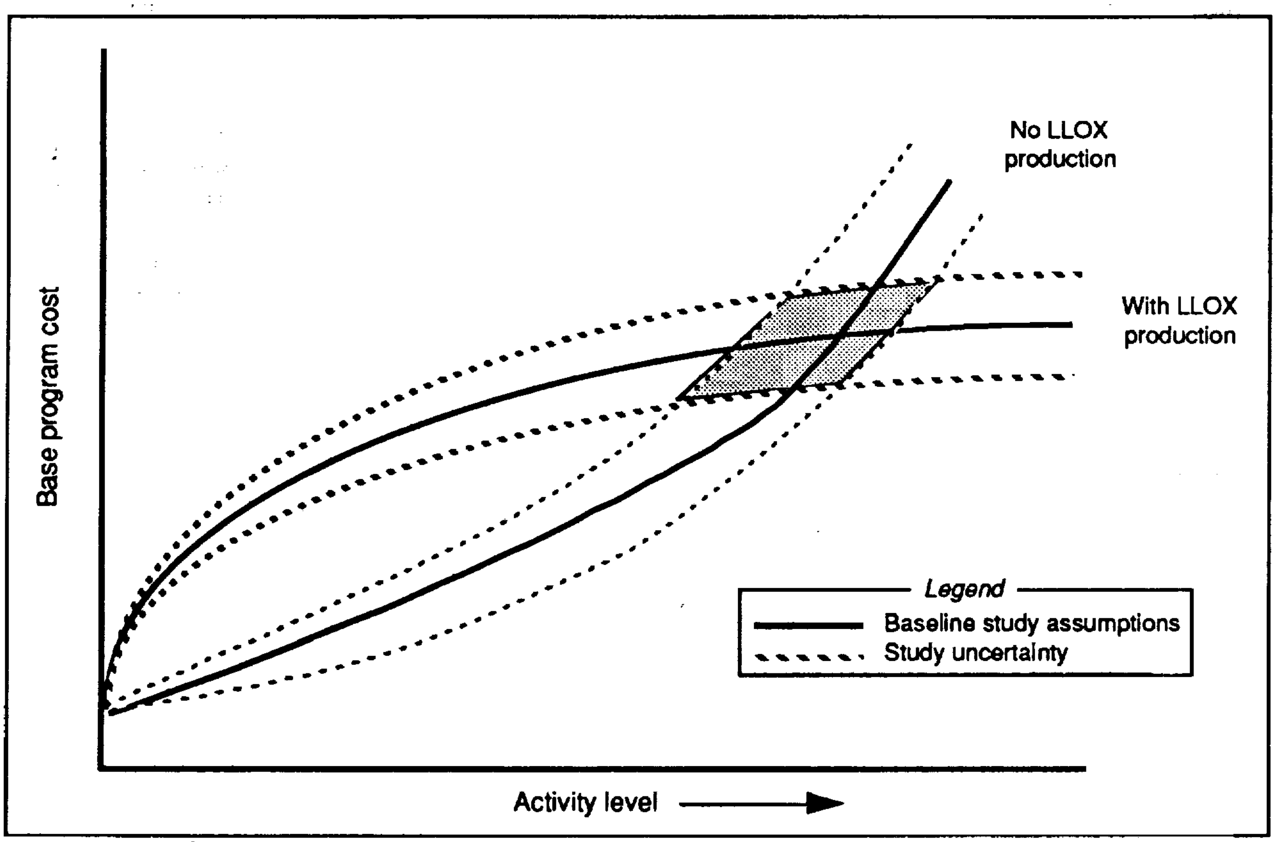


Figure 6.6-1.- Relationship of lunar outpost activity level to LLOX production.

simplify the problem.

For this study, past work on LLOX production was reviewed to derive reasonable ranges for LLOX production performance parameters. Utilization of LLOX was then evaluated to define the lunar outpost activity level at which a reasonable return on investment is achieved. The results of the study are discussed in section 6.6.3.

6.6.1 Overview of LLOX Production Techniques

A literature search was conducted to understand the work that has been done on the potential of LLOX, so that the focus of this trade study could be appropriately narrowed. Areas covered in the search included general overviews of the potential uses for LLOX, oxygen production processes, and the costs and benefits associated with LLOX. Approximately 50 different reports and technical papers were examined.

Two principal conclusions were reached. First, the potential benefit of LLOX depends heavily on the location at which the demand for LLOX occurs. In general, given the same quantitative demand for LLOX at two different locations, for example LEO and LLO, the potential for a return on investment is higher for the location closer to the lunar surface. However, the demand for LLOX in a typical lunar outpost program increases as location increases in distance from the Moon. LEO is farther from the Moon than LLO, but the quantitative demand for LLOX there is greater, thereby allowing economies of scale in producing LLOX. The majority of articles covering LLOX production examine the potential of exporting LLOX to LEO for use in activities that would arise from a space transportation node located there (e.g., human missions to Mars, robotic GEO missions and planetary probes, and support of Strategic Defense Initiative (SDI)). These articles all came to generally the same conclusion: to obtain a reasonable mass payback ratio (i.e., mass to LEO with LLOX/mass to LEO without LLOX < 1.0) or an economic return on investment, the demand for lunar oxygen in LEO had to be very large (more than 1,000 t/year). As a result, in order to narrow the scope of this trade study, the analysis focused on using LLOX to satisfy demand between the Moon and LLO only, in support of a typical lunar outpost. No LLOX was exported to LEO, nor was LLOX used to support human missions to Mars.

The second conclusion involved the transportation architecture used to support a lunar outpost program. The most frequently discussed chemical propulsion systems in the LLOX production studies used oxygen as the oxidizer and hydrogen as the fuel. Using LLOX in these propulsion systems still requires that hydrogen be brought from Earth; unless sufficient quantities can be produced on the Moon. Two possibilities exist for hydrogen: water ice at the permanently shadowed regions of the lunar poles and hydrogen emplanted in the regolith by the solar wind. Obtaining hydrogen from water ice would require that the outpost be placed at one of the lunar poles, which would cause some operational changes compared with an equatorial outpost. However, the presence of water ice is only theorized; no direct evidence has ever been obtained. Extracting hydrogen emplanted by the solar wind is difficult, because hydrogen is scarce in the lunar regolith (50 ppm), and very large quantities of regolith have to be processed. This may require a separate process to extract the hydrogen, since the amount of hydrogen produced in the typical LLOX production process is far less than that required for propulsion systems. Other propellant combinations, such as oxygen and aluminum, or oxygen and silicon, have been considered as alternatives to oxygen and hydrogen. The motivation for using these alternatives is the desire to produce all lunar vehicle propellants from lunar materials, thereby decreasing the logistics required from Earth. However, these "metal" engines have dramatically lower specific impulses (270 seconds versus 480 seconds for advanced LOX/LH_2), requiring larger quantities of LLOX to achieve the same objectives. In addition, the propulsion system designs for these alternative "metal" engines are only in the conceptual stages, and many problems still must be resolved. Due to these uncertainties and in order to focus on the potential benefits of LLOX, only chemical propulsion systems using oxygen and terrestrial hydrogen were considered.

Numerous proposed processes can produce oxygen from lunar materials. Table 6.6.1-I presents some of the pertinent parameters of several representative LLOX production techniques. However, the degree to which these processes are understood varies. Only one process, hydrogen reduction of ilmenite, has been studied in some detail, and many of the other processes are only at the conceptual stage. Nevertheless, a base of knowledge allows the generalities of LLOX production to be described. Table 6.6.1-I shows that the various processes have widely different oxygen production efficiencies in terms of the theoretical and estimated actual oxygen produced per metric ton of mined regolith. The production efficiencies vary by a factor of more than 100. These variations are based on the composition of the lunar regolith and the thermodynamics of the production processes.

Plant masses and power requirements are not as well understood as mining requirements. The existing data indicate that the estimated plant masses and power requirements vary by factors of six. However, because these variations are poorly understood, the real differences will become better defined as further studies are completed. Similar uncertainties in the reactant resup-

TABLE 6.6.1-I.- REPRESENTATIVE LLOX PRODUCTION PARAMETERS

Technique	Theoretical efficiency (%)*	Theoretical efficiency (t/t)**	Estimated efficiency*** (t/t)**	Metals
Low yield				
Hydrogen reduction of ilmenite****	1.8	127	327	Fe
Carbothermal reduction of ilmenite****	1.8	127	160	Fe
Medium yield				
Hydrogen sulfide reduction	23	9.9	25	Mg, Fe, Ca
Molten silicate electrolysis	41	5.6	6.2	Si, Fe
High yield				
Molten salt electrolysis	100	2.3	2.9	Si, Mg, Fe Ca, Al, Ti
Fluorine exchange	100	2.3	2.9	Si, Mg, Fe Ca, Al, Ti
Hydrofluoric acid leaching	100	2.3	2.9	Si, Mg, Fe Ca, Al, Ti
Sodium hydroxide electrolysis	100	2.3	2.9	Si, Mg, Fe Ca, Al, Ti

* % of available LLOX (44%) in typical regolith
** t of regolith processed/t of LLOX
*** Only the metals that occur in lunar materials in >1% abundances are listed; i.e., Si, Mg, Fe, Ca, Al, and Ti
**** Assuming a maximum of 7.5 % free ilmenite in high titanium mare regolith

ply, spare parts requirements, and crew requirements for set-up, production, and maintenance exist for all the proposed production techniques. For the purposes of this report, the existing information does allow reasonable limits to be set for these propellant processing performance parameters.

6.6.2 Study Approach

The basic approach used in this study to assess the potential benefits of LLOX was to establish a realistic reference lunar outpost scenario that does not incorporate the production of LLOX. This reference outpost was then compared to the same outpost with LLOX production equipment and its associated overhead and infrastructure included. Total costs for the two programs were then determined on a year by year basis and the cumulative program costs were compared. The intersection of the two curves, if one occurs, designates the time to a return on investment. Because of the relative uncertainty in the various parameters (discussed in section 6.6.1), the study was conducted parametrically. Thus, rather than a single point defining the time to a return on investment, a bounded region is outlined.

Since the objective of the trade study is to define the

activity level at which a reasonable return on investment is achieved, various levels of activity had to be examined. However, the activity level is a function of the type of lunar outpost program. Two major approaches to modelling a lunar outpost can be used to determine the effects of LLOX on the system. In the first approach, the modelling tracks the evolution of an outpost development scenario, with and without LLOX production, starting with the first flight in the outpost development program and stopping at a designated point in the development sequence. This approach has the advantage of being highly accurate in that it takes into account all the details of the full development sequences of the outpost. However, the activity level (i.e., the level of LLOX demand) is not constant, making the results highly dependent on the characteristics of the evolutionary sequence chosen, in particular the outpost's rate of expansion and the point at which LLOX is introduced into the program.

In the second approach, the modelling tracks only the differential effects of adding LLOX production to an outpost that has attained a "steady-state" level of activity. By considering only the differential effects, the results are independent of the evolutionary path of the outpost. By examining several different levels of steady-state activity, the level at which a reasonable return on investment is achieved can be determined. This approach reduces the amount of time needed for the study because representative levels of activity for an evolutionary outpost can be selected without burdening the study with how the outpost got to that level of activity. Although the differential effects of LLOX production are accurately modelled, the absolute impacts are less precise because the costs of the outpost buildup to the steady state are not tracked via detailed modelling, but they are calculated directly from the costs of the materials at the steady-state outpost and their transportation costs.

After evaluating the advantages and disadvantages of these two approaches, the second was chosen because the significant savings in study time it offers outweighs any slight imprecisions in the final results. Additionally, three levels of steady-state outpost activity were chosen to bracket the problem and reflect a reasonable spectrum similar in scope to that considered by the Office of Exploration in lunar-related case study analyses. These three levels are represented by outpost crew sizes of 4, 12, and 36. The assumptions relating to each level are described in section 6.6.2.2.

6.6.2.1 Measures of "Leverage"

In this study, leverage means any enhancement of the lunar outpost program by the utilization of lunar-produced oxygen. This enhancement may be measured directly, by a number of tangible quantities, or indirectly, by a number of intangible quantities.

Any enhancement of a lunar outpost by the utilization of LLOX can be numerically measured by a number of different, but not necessarily independent, parameters; these are considered the tangible quantities. For example, mass delivered to LEO and the lunar surface is a direct measure of the effectiveness of a lunar outpost scenario, and it can be used as a measure of the leverage LLOX has on the program. If the mass through LEO is held constant as a result of Earth-to-LEO launch limitations or program cost limitations, then the ratio ((mass to the surface)/(mass to LEO)) would increase if LLOX has a positive effect. The outpost could grow even if the mass through LEO stayed constant. Alternatively, if the level of activity were held constant, LLOX leverage would mean that less mass would have to be launched from Earth to maintain that level.

The cost of the lunar outpost program is also a direct measure of the leverage of LLOX. If LLOX has a positive effect, the cost of the program would decrease. At a constant annual investment, LLOX leverage would result in expanded growth of the outpost's capabilities. Alternatively, if the level of activity were held constant, a LLOX leverage would mean that the cost of maintaining that level would decrease.

As the LLOX facility is established on the Moon, the lunar and cislunar infrastructure will change. The addition of LLOX plants, mining and beneficiation equipment, electrical power plants, and other associated overhead will increase the infrastructure of the lunar outpost. These increases may be offset by decreases in the infrastructure needed to support the transportation system. As LLOX is used in the landers and perhaps in the space transfer vehicles, fewer transfer vehicle flights may be needed to bring the same payload to the lunar surface. These lower flight rates would decrease both the size of the vehicle servicing facilities at the LEO node and the number of transfer vehicles required to support the outpost. Also, since LLOX can be used in the life support systems, the ECLSS could be designed to rely more on LLOX to achieve closure of the air and water loops, instead of relying totally on physical/chemical systems. Thus, the net change in the lunar outpost/cislunar infrastructure will also indicate the degree of leverage that LLOX has on the system.

Another tangible measure is the required level of crew involvement. As the LLOX facility is established, the personnel requirements at the outpost and at the LEO node will change. The addition of the LLOX plants, mining and beneficiation equipment, and electrical power plants may increase the need for crew members at the outpost. These increases may be offset by decreases in the need for crew to support the transporta-

tion system. Also, since LLOX is used in the life-support systems, the crew support of the ECLSS may decrease. The net change in the lunar outpost/cislunar crew requirements will also indicate the degree of leverage that LLOX has on the system.

Because LLOX is used in the transportation and life support systems, the payloads delivered to LEO and the lunar surface may consist of more hardware (scientific, resource utilization, and crew support equipment), and less oxygen propellant and life-support oxygen and water (89 percent oxygen). The mass ratio (hardware/propellant) will increase if LLOX makes the transportation and life support systems more efficient.

In addition, the degree of enhancement of the lunar outpost by the utilization of LLOX can be evaluated by a number of different, non-numerical quantities; these are termed the intangible quantities. For example, if LLOX production can cover all life support oxygen requirements and can provide the lunar and cislunar transportation system with oxygen propellant, the outpost may become more self-sufficient and more independent of Earth-derived support. The self-sufficiency would be a direct result of not needing terrestrial oxygen for propellant and life support. Indirectly, self-sufficiency would result from the growth of the outpost and the expanded presence of humans on the Moon, as discussed in the following paragraphs.

Outpost synergism is another intangible measure of leverage. Development of a LLOX facility requires the emplacement of a number of support elements, such as electrical power supplies, regolith mining and sorting equipment, and transportation equipment. Once this equipment is available, it can be used to expand the outpost by supporting additional resource utilization activities, such as making cast and/or sintered basalt for construction, obtaining volatiles (e.g., H_2, N_2 and helium-3) by vacuum pyrolysis of the lunar fines, etc. In addition, the production of LLOX yields one or more metals (mainly Si, Mg, Fe, Ca, Al and Ti) as by-products, depending on the production process. These metals can be refined and used to make additional construction elements for use at the outpost and for export. Finally, the LLOX may expand the outpost's capability to support science activities. For example, the exploration of the Moon can be extended to the global scale by providing LLOX propellants for manned ballistic research vehicles and LLOX for the life support systems of the long-distance surface rovers.

The effect on the outpost's growth potential is also a significant criterion. If LLOX reduces the operational costs of the outpost and/or increases its capabilities, its growth potential will be enhanced. An expanded capability to bring more mass and/or higher quality mass to the lunar surface would allow additional scientific equipment to be brought to the Moon (e.g., life sciences laboratories, ballistic research vehicles, and astronomical telescopes). The resource utilization facilities could be expanded to start making construction elements from lunar materials to build habitation and laboratory facilities. These and similar capabilities may allow off-outpost, man-tended facilities to be established in areas of high scientific interest (e.g., an exploration site in Mare Orientale or a farside astronomical facility).

Human presence on the Moon and in cislunar space may be increased if LLOX reduces the operational costs of a lunar outpost and/or increases the capabilities of the outpost. Additional personnel may be required to operate and maintain the LLOX facilities, the mining equipment, and any related lunar resource utilization facilities. Using lunar oxygen in the transportation system may require additional personnel to maintain the transportation vehicles and prepare and fuel them for launch. These and similar activities not only would increase the number of humans in space and on the Moon, but also would expand the areas in which experience in working in space and on the Moon is gained.

6.6.2.2 Study Assumptions

Several assumptions were made in the course of the study to simplify the analysis or because of the lack of maturity of the available data. These assumptions were made in two areas: engineering and cost.

The engineering assumptions made include:

a. Only chemical propulsion for transportation systems

b. Lunar transfer vehicle used for LEO to LLO (300-km circular orbit) transport

c. Lunar excursion vehicle used for LLO to lunar surface transport

d. All engines use LOX/LH_2 propulsion with Isp=482 sec and O/F=5.5

e. All vehicles fully reusable

f. Transfer vehicles fueled and serviced in LEO at Space Station Freedom

g. Excursion vehicles serviced on the lunar surface and fueled in LLO

h. In scenarios using LLOX, excursion vehicle is filled with LLOX on the lunar surface and with terrestrial hydrogen in LLO

i. No LLOX used in transfer vehicles

j. Excursion vehicles sized to deliver 20 t of payload to the lunar surface (when operating in the cargo mode) consistent with the excursion vehicle in the Lunar

Evolution case study

k. No LLO facility.

The cost assumptions include:

a. Development and production cost estimated using NASA JSC's Advanced Mission Cost Model (see section 6.6.2.3)

b. ETO transportation costs use Shuttle-C capability of 71 t of cargo to LEO for a fixed amount per flight, independent of flight rate

c. High value for ETO transportation cost is three times baseline

d. Low value for ETO transportation cost is one-third of baseline

To evaluate the sensitivity of the modelling results to the steady state level of activity, the generic baseline scenario was evaluated at three different activity levels. Each of these activity levels, which are discussed below, is represented by different crew sizes and outpost capabilities.

1. **Four-person outpost:** The level of activity of the four-person outpost defines the minimum considered for purposes of the study. The life support system closure is equivalent to that of Space Station Freedom; i.e., 90 percent closure of water and oxygen using physical-chemical systems. Lunar resource utilization experiments and production are limited to those related to LLOX. Global lunar science and prospecting are limited to one ballistic mission per year to a remote site for a three-person crew for a 2-week period. Lunar science support equipment (sample collection and analysis equipment, portable geophysical equipment, etc.) is considered to be in steady state with resupply only being brought to the outpost. The astronomical equipment is increased by one additional telescope every 3 years. Life sciences research equipment is considered to be in steady state with resupply only being brought to the outpost. The outpost requires one piloted and one cargo lunar excursion vehicle flight per year to deliver the required payload to the lunar surface.

2. **12-person outpost:** The level of activity of the 12-person outpost defines the median for the purposes of the study. The life support system uses advanced physical-chemical and bioregenerative systems, including the utilization of food growth chambers to provide part of the vegetable foods for the crew. Lunar resources utilization experiments and production include those related to LLOX and simple regolith processing like sintering and casting basalt construction elements and volatile extraction. Global lunar science and prospecting are carried out by three ballistic missions per year to remote sites for three-person crews for 2-week periods. Lunar science support equipment increases at a modest rate each year. The astronomical equipment is increased by one additional telescope every second year. Life sciences research equipment is increased at a modest rate each year.

3. **36-person outpost:** The level of activity of the 36-person outpost defines the maximum for the purposes of the study. The life support system is nearly closed, food growth chambers provide all vegetable foods for the crew, and experimental meat production is underway. Lunar resources utilization production includes LLOX, simple regolith processing (sintering and casting basalt construction elements) and volatile extraction and complex processing (metals separation and refining to make construction elements). Global lunar science and prospecting are carried out by three ballistic missions per year to remote sites for three-person crews for 2-week periods, and by a long-distance, long-duration rover mission once every 3 years. The latter requires ballistic vehicle resupply and crew rotation support flights every 2 months. Lunar science support equipment increases at a modest rate each year. The astronomical equipment is increased by one additional telescope every year. Life sciences research equipment increases at a modest rate each year.

6.6.2.3 Engineering and Cost Model

The computer model used in the study employs a series of interrelated spreadsheets in which the mass, power usage, manpower requirements, cost, etc., of each element used on the lunar surface or as part of the transportation system are contained (see figure 6.6.2-1). The payload capacity, propellant usage, specific impulse (Isp), oxidizer-to-fuel ratio (O/F), and other performance characteristics of the transportation vehicles are also listed, as are the LLOX production parameters for generic processes. Similar data are available in the spreadsheet for different support systems and all other pertinent systems.

Once the lunar outpost elements chosen for the run are identified, the model calculates all the parameters (mass delivered to the lunar surface, oxygen usage, power usage, etc.) needed to define the support requirements (number of power and LLOX plants, number of space transfer vehicle and lander flights, etc.) of the crew and the initial elements. The model then adds these support elements to the initial list of elements and iteratively converges on the final results.

Total outpost program costs were determined by computing the development and production costs for each element included in both the LLOX and non-LLOX scenarios and adding these totals to the transportation costs

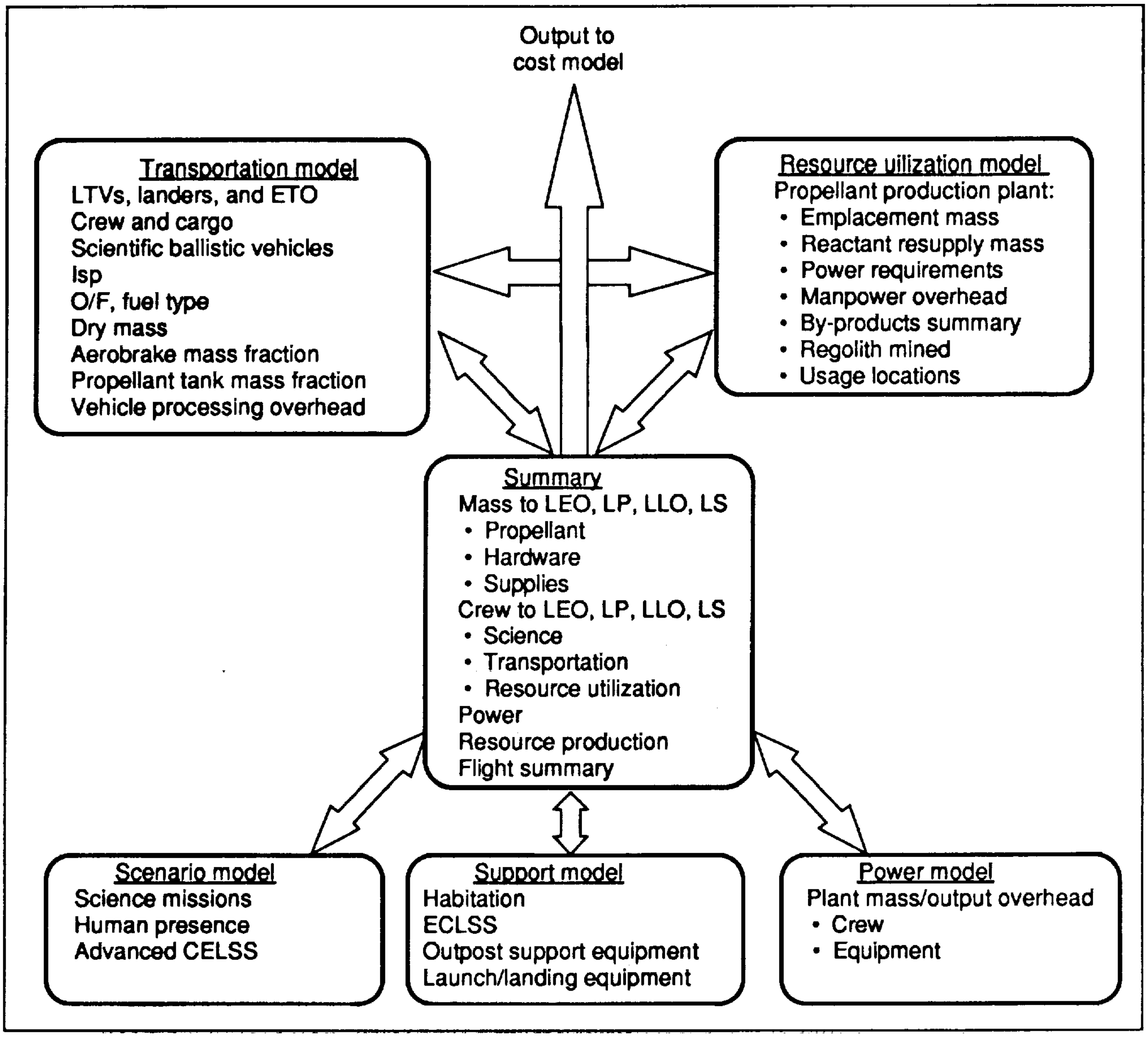


Figure 6.6.2-1.- Engineering model.

involved in delivering these elements to the lunar surface. Using the masses of each of the elements and the transfer and excursion vehicle transportation system, the total transportation requirements through LEO were calculated in the model. With the Shuttle-C as the baseline Earth-to-orbit system for delivering cargo to LEO, transportation costs for each scenario were determined by calculating the required number of Shuttle-C launches. No costs were included for operations.

The modelling is performed by running the baseline scenario initially without and then with LLOX. In order to allow accurate comparisons between the LLOX and non-LLOX runs, adjustments have to be made to account for differences in the ways certain outpost objectives are achieved. For example, a ballistic research vehicle is used in the baseline scenarios to transfer lunar science crews from the outpost to exploration sites over the lunar globe. In order to keep the scenarios equivalent and realistic, these flights are run from low-polar lunar orbit in the non-LLOX baseline rather than from the outpost. Aside from such equivalency adjustments, the only parameters varied in the baseline are those directly related to the production of oxygen.

6.6.3 Study Results

This section summarizes the results obtained during this year's study. Included is a description of the LLOX plant sizing and support requirements. The trade study began by focusing on the activity level represented by a steady-state, four-person lunar outpost. This narrowed the scope of the parameters used as inputs to the Engineering and Cost Model. For example, multi-megawatt-class nuclear power is improbable for such a small outpost, as are large inflatable or lunar-derived habitats. By

performing the trades first on the four-person outpost, if a reasonable return on investment were found, the need to continue the study for higher activity levels would be eliminated.

6.6.3.1 LLOX Production Plant Sizing and Support Requirements

The lunar propellant leveraging trade study was performed parametrically. The parametric values of each of the critical parameters used in the study are discussed below.

Earlier trade studies have shown that the LLOX propellant requirements of a lunar outpost are generally satisfied by one or more unit plants, each of which has a production capacity of about 100 t/yr. This result is used as a baseline for this study. However, in order to accommodate yearly LLOX requirements that are not integer multiples of the baseline 100 t/yr production rate without introducing over-production errors into the analyses, the number of production plants, power units, etc., used is allowed to vary continuously rather than incrementally. For example, if the LLOX utilization requirements are 230 t/yr, the model will use 2.3 plants to produce this LLOX, rather than three plants with a 70 t/yr over-production. Clearly, a single plant could be designed to produce 230 t/yr, rather than transporting three separate 100 t/yr plants to the Moon.

The variations and uncertainties of the LLOX production processes are taken into account in the study by considering three generic LLOX plants and support equipment (diggers, conveyors, etc.). Each plant is designed for 100 t LLOX annual output. The requirements associated with each of these plants are summarized in table 6.6.3-I.

The power plants required to support the LLOX facilities (and the other outpost facilities) are adjusted continuously to meet the requirements of each of the scenarios studied. The power units considered in the study are treated parametrically in five classes of increasing efficiency and are summarized in table 6.6.3-II. The particular power plant used in the analysis depends upon the level of demand.

The operational crew support needed to run the facilities on the Moon is estimated to be 0.3, 1, and 3 crew per 100 t of LLOX produced per year. The lower bound represents an Earth-operated plant with lunar-crew maintenance; the upper bound represents a facility operated by lunar outpost crew only.

Three classes of crew habitation facilities are considered in the study. The first class is based on Space Station Freedom-derived modules, airlocks, and nodes, with a mass of 6.8 t per person. The second class of facility is based on inflatable structures, which have a mass of 4 t per person. The third class is based on facilities derived from lunar resources, with a mass of 2.5 t per person. This mass is simply the mass of the Earth-derived equipment needed to outfit the facility, the shell of which is made from lunar-derived materials. Volume VI of this Exploration Studies Technical Report contains the results of trade studies to define the parameters described above.

TABLE 6.6.3-I.- GENERIC LLOX PRODUCTION PLANT STUDY PARAMETER VALUES

Parameter	Low value	Median value	High value
Emplacement mass (t)	10	25	65
Electrical power requirements (kW)	300	750	1,800
Reactant resupply (t/yr)	0.1	0.5	3
Spare parts mass (% of plant mass)	3	10	30
Crew support man-years/plant	0.3	1.0	3.0

TABLE 6.6.3-II.- CLASSES OF POWER SYSTEMS

	1	2	3	4 *	5
Power rating (kg/kW)	400	50	33	25	10
Type	PV/RFC	SP-100 class	PV	nuclear reactor with Stirling engines	advanced multi-megawatt nuclear reactor

* Based on NASA-Lewis Research Center 825 kW nuclear reactor concept

6.6.3.2 Integrated Results

The results presented here are based on the level of activity for the four-person lunar outpost. SP-100 class nuclear power is to supply all power needs on the lunar surface. Space Station Freedom-derived habitation modules are used for crew support, and air and water loops for life support are 90 percent closed.

Figure 6.6.3-1 presents the overall results of the cost analysis using the reference cost estimates. The three curves for LLOX production illustrate the bounds of the parametric problem. The "No LLOX" case uses the median values for all the study parameters. The "LLOX Low" case represents the result of using the lower bounds for all the engineering parameters (i.e., LLOX plant mass, LLOX power demands, reactant resupply, LLOX plant spares, and LLOX plant crew support). The median and high cases represent the associated median and high values being used for the study parameters. As figure 6.6.3-1 illustrates, a return on investment occurs with a four-person lunar outpost level of activity in all but the worst case. In fact, a return on investment is achieved after only 4 to 9 years in the low and median cases. Further, the reduction in total program costs after a 20-year period could be as high as 15 to 25 percent. Figure 6.6.3-2 presents the same data using mass to LEO as the scale for "return on investment" (mass payback).

Figure 6.6.3-3 shows individual sensitivities to variations in the major LLOX production parameters represented by the number of years to a positive return on the investment in LLOX production. It is interesting to note that the number of years to payback of overall costs is almost twice the number of years to payback of overall mass to LEO. Previous studies have used mass to LEO as the only discriminator of the potential for LLOX utilization; however, this only holds if transportation costs dominate total program costs. The results shown in figure 6.6.3-2 show costs other than transportation to contribute significantly to total program costs. No reasonable time to payback was achieved using high parameter values for either overall costs or overall mass to LEO. Also shown in figure 6.6.3-3 are the sensitivities of parameter values for the LLOX plant size, the supporting power system, reactant resupply, plant spares, and crew support. Each of the systems was analyzed holding all other parameters to their median values. A discussion of parameter value effects on the time to a positive return on investment follows.

1. LLOX plant size: The time to payback shows only a mild sensitivity to the LLOX plant size. The LLOX plant size was estimated at 10, 25, and 65 metric tons for the low, median, and high values respectively.

2. Power system support: The most sensitive system to

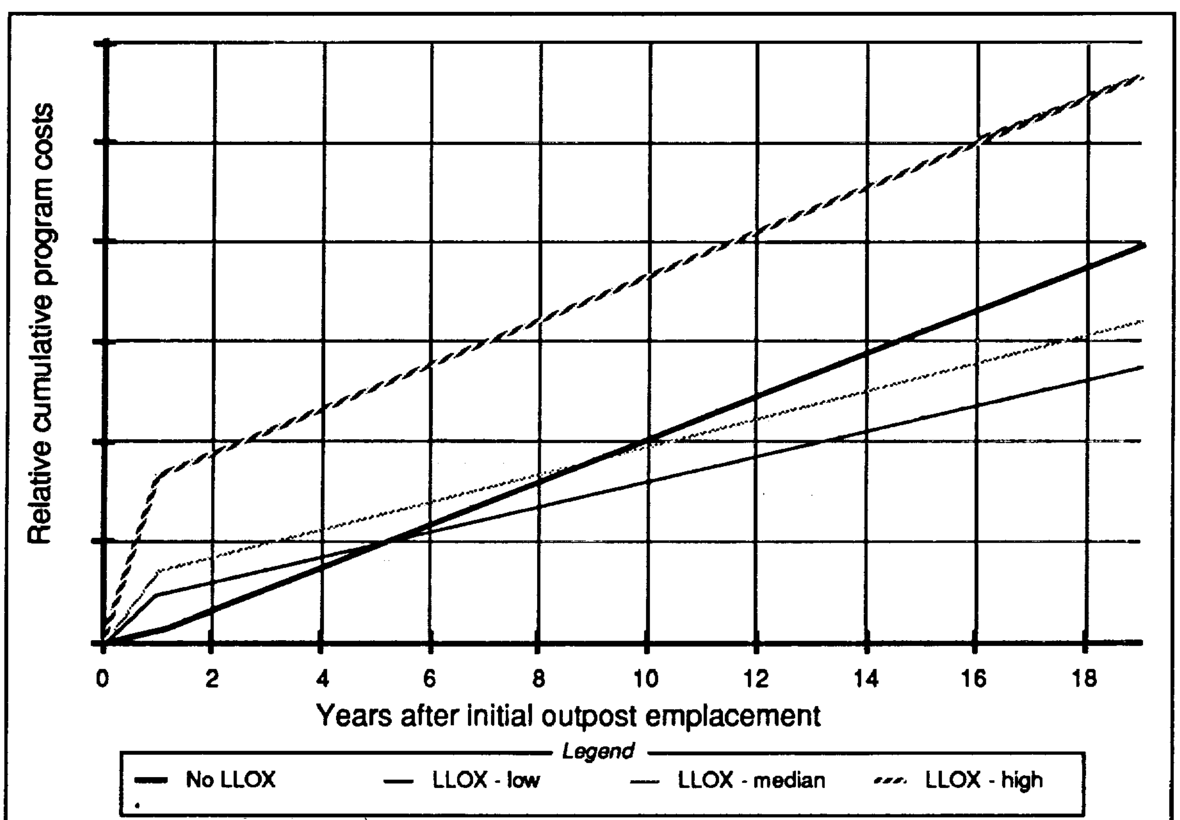


Figure 6.6.3-1.- Lunar propellant leveraging trade study overall sensitivity results: cost.

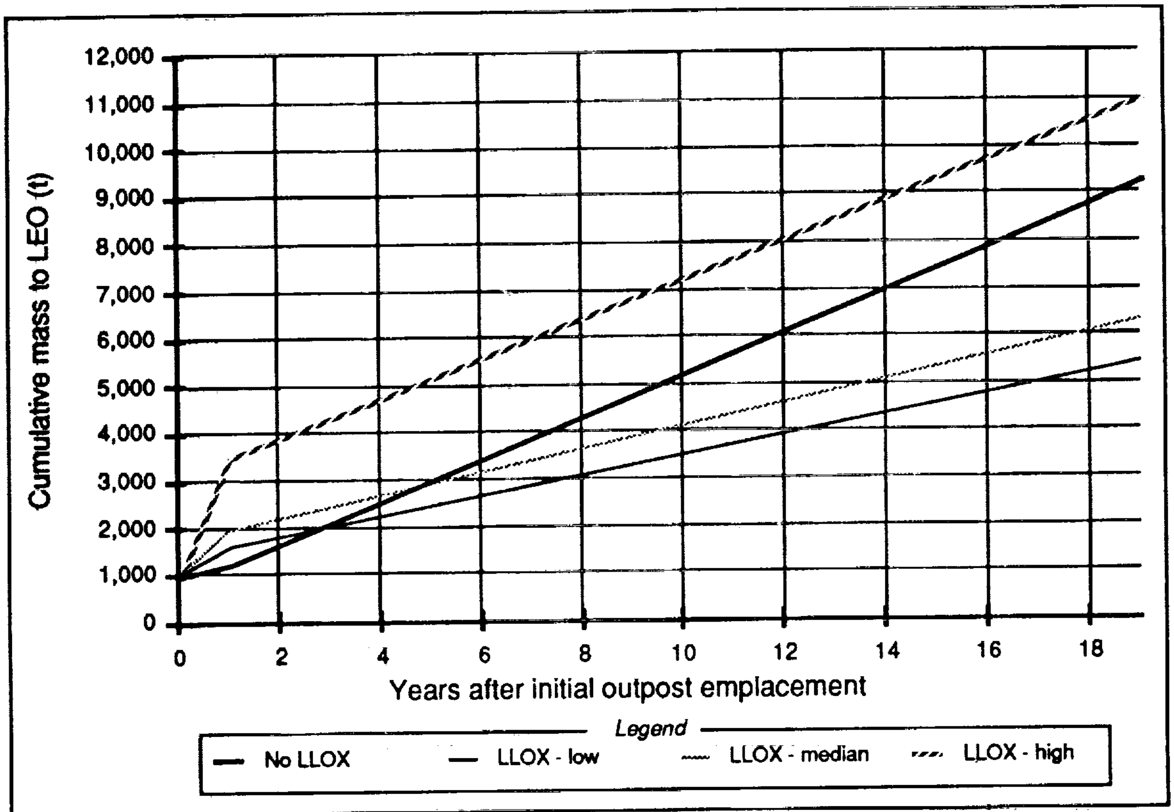


Figure 6.6.3-2.- Lunar propellant leveraging trade study overall sensitivity results: mass.

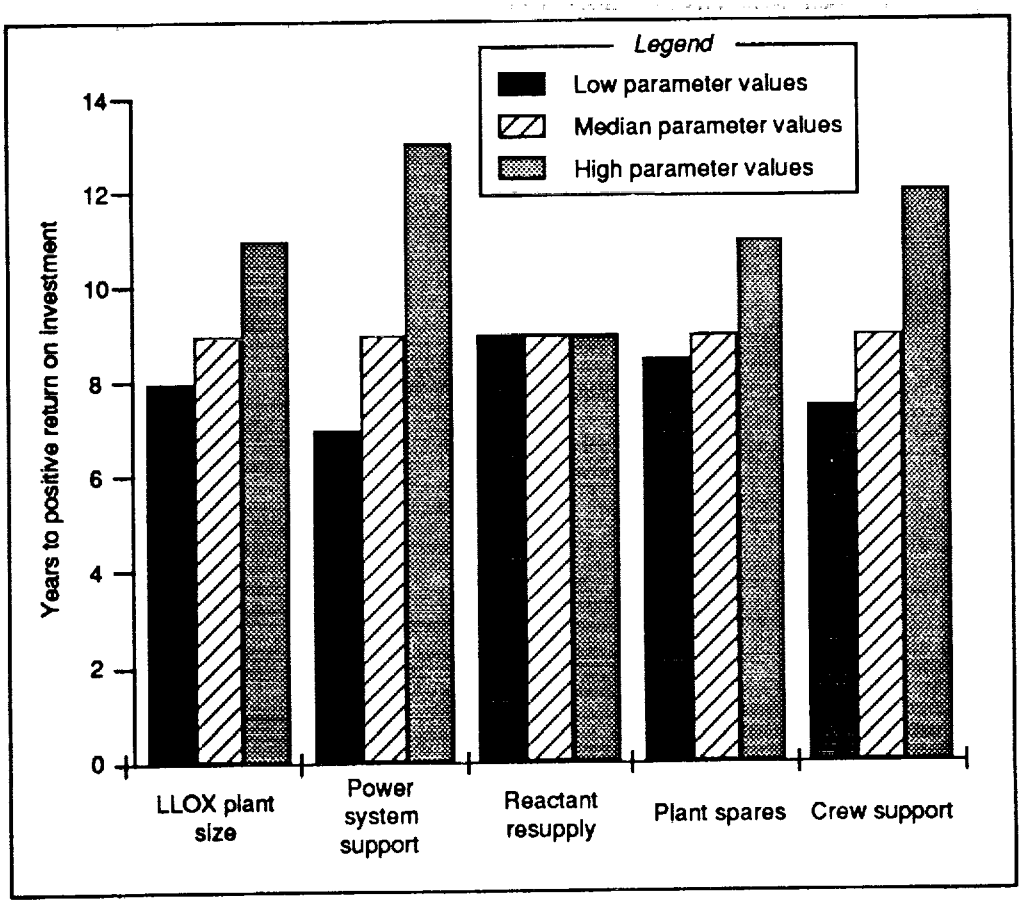


Figure 6.6.3-3.- Time to positive return on investment showing individual sensitivities to variation in major LLOX production parameters.

parameter value estimates seems to be the supporting power system, which shows anywhere between 7 and 13 years to payback depending on the representative parameter values. Electrical power requirements for the study were estimated at 300, 750, and 1,800 kW for the low, median, and high values respectively. Only electrical power was used; no thermal power from the SP-100 was used in LLOX production.

3. **Reactant resupply:** Time to payback does not show sensitivity to variations in reactant resupply parameters. For LLOX production via hydrogen reduction of ilmenite, almost all the reactant is estimated to be recoverable, and resupply is minimal relative to other support requirements. For this study, reactant resupply was estimated at 0.1, 0.5, and 3 t per year per plant for the low, median, and high values respectively.

4. **Plant spares:** Time to payback shows a mild sensitivity to requirements for plant spares. Spare part mass in this study was estimated to be 3, 10, and 30 percent of total LLOX plant mass for the low, median, and high values respectively.

5. **Crew support:** Time to payback shows a strong sensitivity to the need for crew support of the LLOX plant ranging from about 7 to 12 years. This is due to the high costs associated with crew support on the lunar surface. For the low parameter value, the plant is operated from Earth and shows a relatively quick time to payback. As plant support becomes lunar-based, crew life support requirements contribute significantly to program costs.

Figure 6.6.3-4 presents the results of holding the engineering parameters to their median values, but varying the transportation and development/production costs in order to account for uncertainties in the cost estimates. Note that in all cases, a return on investment was still achieved. However, the results indicate that if Earth-to-orbit transportation costs are high relative to the development/production costs of the LLOX hardware, LLOX has a very high leveraging effect. If the LLOX hardware development/production costs dominate the ETO transportation costs, then the positive impact of LLOX on the outpost is reduced, but not eliminated.

The main result of the study to date is that the utilization of LLOX in the life support system, for lunar science missions, and for the lunar surface to LLO leg of the transportation system of a minimum outpost with a

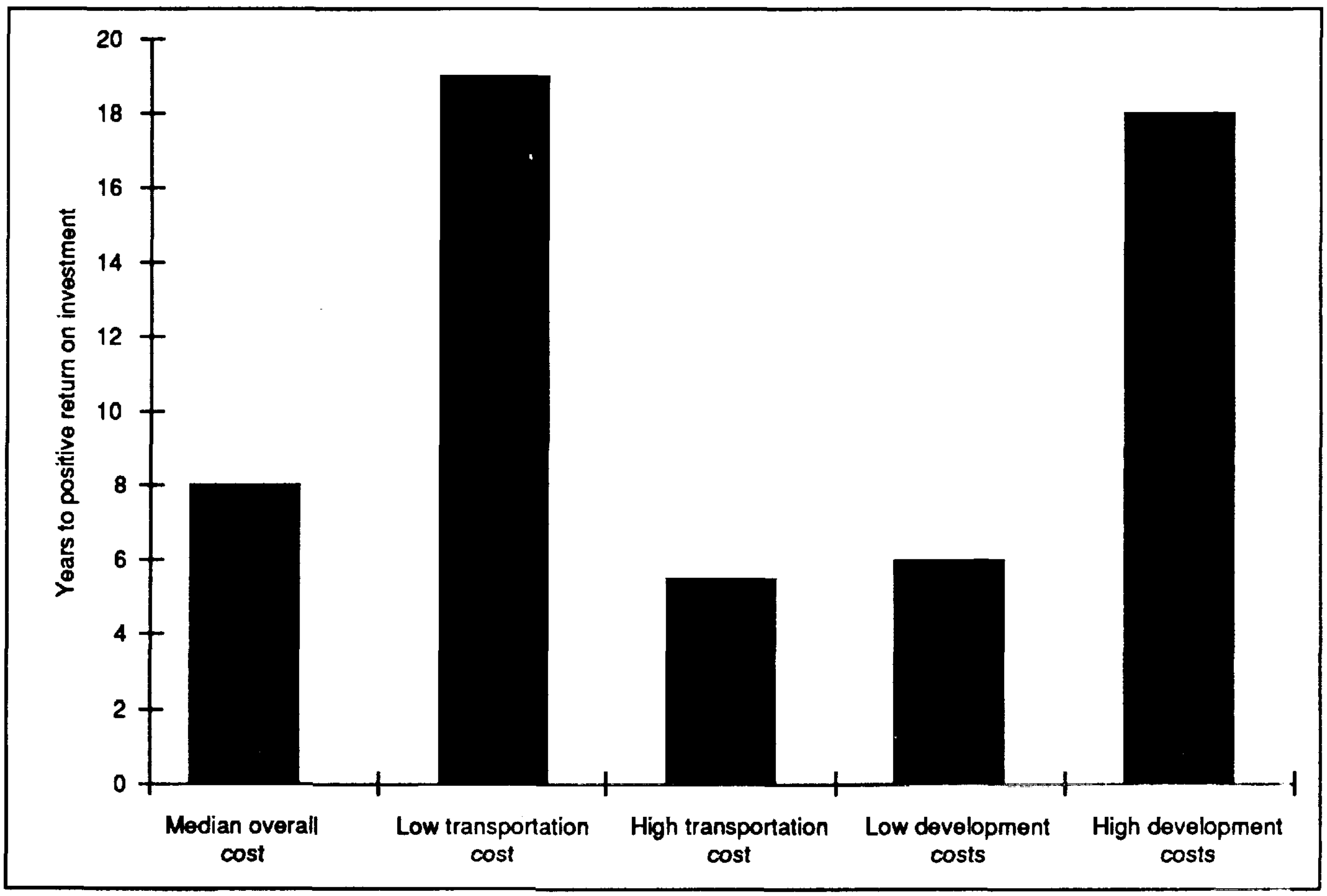


Figure 6.6.3-4.- Time to positive return on investment holding engineering parameters to median values showing sensitivity to cost estimates.

crew as small as four results in significant savings after a time period characteristically less than 10 years. Only in the extreme case, where all the input parameters are at their maximum values, does LLOX show a negative impact on the base costs.

Many simplifying assumptions were made during the course of the study; these must be assessed in future analysis. In particular, no operations costs were accounted for in the cost model analysis. Differences in operations costs between the LLOX and non-LLOX outposts would apparently fall in three areas: (1) ETO transportation operations costs, (2) transfer vehicle operations costs, and (3) LLOX production plant operations costs. The first two operations costs would probably be lower for the LLOX versus non-LLOX case and, therefore, not including the costs is a conservative assumption. The operations costs associated with the LLOX production plant will obviously be greater for the LLOX-supported outpost. However, analysis performed during the course of this study (see section 4.5 of Volume VI) indicates that the plants would be virtually autonomous, requiring minimal Earth teleoperation support and outpost crew support for plant maintenance only. Thus, neglecting these costs in the analysis should not have a dramatic impact on the results.

The greatest area of uncertainty in the results is due to the poor definition of the LLOX production processes and equipment. In particular, the size, complexity, and support requirements must be examined in greater detail. Additionally, the costing analysis done for the LLOX production plants and support equipment reflects an aerospace/NASA bias for man-rated systems. Terrestrial mining costing experience indicates equipment may be as much as one to two orders of magnitude less expensive than the cost estimates used in this study. If this is the case, the potential for a return on investment would increase substantially.

Finally, in situ resource utilization must be more completely integrated in the outpost development in future analysis. Producing LLOX creates many potentially useful by-products that can significantly alter outpost development and logistical resupply strategies.

6.7 LAUNCH/ON-ORBIT PROCESSING

6.7.1 Objectives

The objectives of this controlled trade study were to analyze the trades associated with launching, assembling, and servicing a space transfer vehicle (STV) and to derive an effective blend of ETO launch capability, STV design, and ground and space processing capabilities to support a given set of mission requirements. Inherent in this objective is the need to identify the tasks to be performed, both in orbit and on the ground, to support STV assembly, checkout, and servicing, and to establish requirements for supporting launches from Earth to LEO. Once tasks have been identified, trade studies may be completed to identify the optimum division between tasks to be accomplished on the ground and those to be completed on-orbit.

6.7.2 Approach

Descriptions of five key elements are necessary for the analysis process: (1) mission requirements, (2) space transfer vehicles, (3) ETO launch vehicles, (4) on-orbit operations, and (5) ground operations. Each of these elements must be evaluated in relation to the others. The study flow, as discussed in the following paragraphs, is illustrated in figure 6.7.2-1.

Given the mission requirements (OEXP case studies) and STV concepts, a mission model was created that represents the total delivery requirements (hardware and propellants) to support the stated mission. Payloads in this model were then manifested, defining the items delivered to LEO on each launch. The manifest was driven by the type of launch vehicle used, since the different capabilities of launch vehicles result in different constraints on the number, weight, and volume of the items that can be delivered simultaneously.

Once the preliminary manifest was identified, the on-orbit facilities required to support this scenario were also identified. The number of additional launches required to provide the facilities were determined and then incorporated into the manifest. The effect is to increase the LEO delivery requirements to include not only the STVs and payloads required for the mission, but also the facilities required to support them in LEO. On-orbit facility requirements are driven by the number and type of elements that must be processed in orbit; these elements, in turn, depend on the manifesting constraints (i.e., launch vehicle capability) that impact the extent to which they can be assembled on the ground.

With the total delivery requirements identified, a final manifesting for each case could be established. At this point, the ground and on-orbit operations requirements could be assessed and the manpower/cost requirements could be defined. This procedure was iterated on to find a solution that effectively divides the required ground and on-orbit operations.

Ground operations requirements are derived from the ETO vehicle type and the total yearly launch-rate requirements; the driver is the maximum launch rate predicted over the life of the program. The ground facilities must be sized for this capacity, which determines the nonrecurring investment. The recurring costs are generated

as a function of the yearly launch rate.

After each ETO delivery concept was fully defined, each combination of options was evaluated against its corresponding mission model to generate funding profiles that can be used to compare the various options. In addition, data were generated for the following parameters:

a. STV dry mass to LEO

b. Propellant mass to LEO

c. Facility mass to LEO

d. Total mass to LEO

e. Number of crew required in LEO

f. EVA requirements

g. Crew on-orbit duration

h. Number of ETO launches

i. Ground facility requirements

j. Ground crew requirements

The options considered for each element of the trade space are shown in table 6.7.2-I, and discussed in the individual sections below.

6.7.3 Results

Results included in this section are based on the lunar and Mars surface element manifests developed to meet

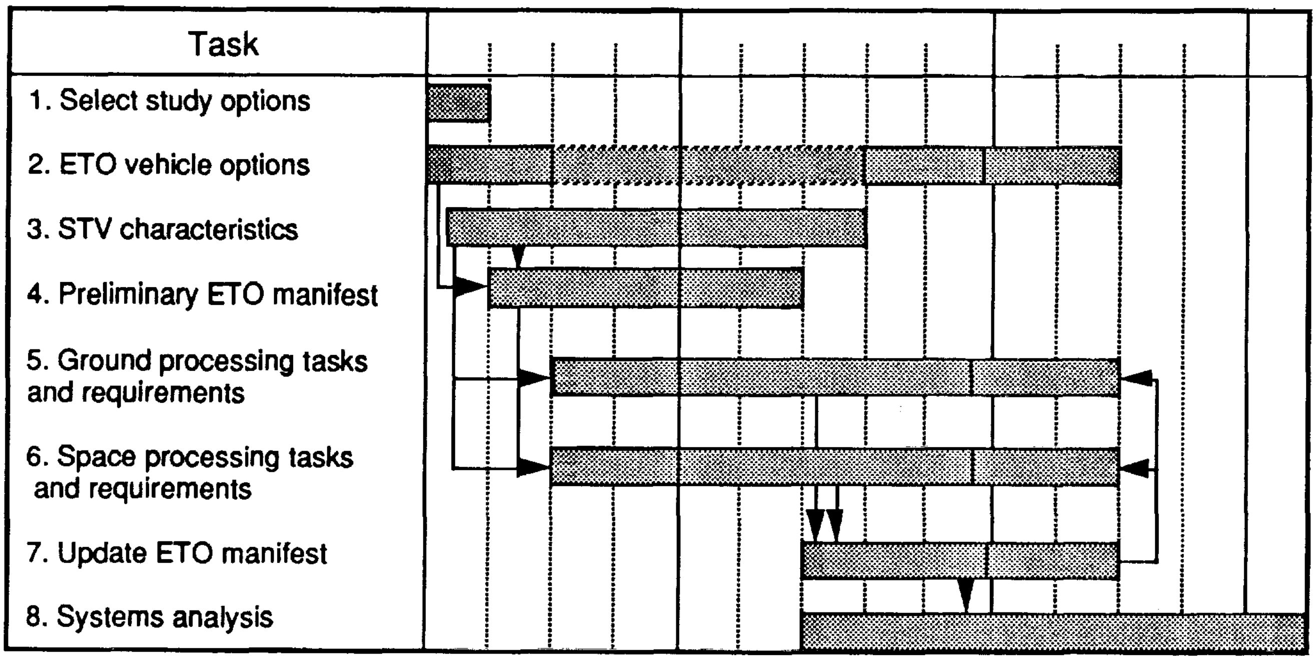


Figure 6.7.2-1.- Launch/on-orbit processing study flow.

TABLE 6.7.2-I.- EXAMPLE TRADE SPACE

Space transfer vehicle	ETO	Ground operations	Space operations	STV mission
Lunar STVs	Shuttle	Process all up	Minor operations	Lunar Evolution
Mars high L/D	Shuttle-C	Component processing	Major component assembly	Mars Evolution
Mars low L/D	Shuttle-Z	Subsystem processing	Subsystem assembly	Mars Expedition
NTR	HLLV			
NEP	ALS			

the mission requirements of the OEXP case studies. Total ETO delivery requirements were established by adding the propellant delivery requirements, derived from the STV definition studies, to the elements included in these manifests. All space transfer vehicle definitions are summarized in section 3 of this document. Additional details can be found in Volume II of this technical report.

Several ETO launch vehicles were considered as poten tial candidates. However, the cargo delivery analysis was limited to the three vehicles shown in figure 6.7.3-1: (1) Shuttle-C, (2) Shuttle-Z, and (3) a new design HLLV. The Space Shuttle, also shown in the figure, was assumed to perform all crew deliveries. When it was found that the driving element in all case studies was propellant delivery (on the order of 70 to 80 percent of total launch stack mass), an alternative, potentially low-cost vehicle for propellant delivery was investigated (figure 6.7.3-2). In all cases analyzed, the most conservative version (lowest performance, highest dry weight, etc.) of the propellant tanker was used, and performance capabilities were based on current Shuttle constraints. In actuality, this vehicle should not be bound by the same constraints, and the resulting performance improvement is yet to be determined.

The following paragraphs summarize the considerations for selection of launch vehicles considered for manifesting respective payloads in each case study. The results described here are based on early case study development and do not reflect the final integrated mission results described in section 3.

Lunar Evolution Class Mission. The annual launch rates required to deliver the Lunar Evolution LEO facility, vehicles, payloads, and propellants to orbit utilizing the Shuttle-C and HLLV are shown in figures 6.7.3-3a and 6.7.3-4a. As can be seen, propellant is the major component of total mass delivered to LEO. Figures 6.7.3-3b and 6.7.3-4b show the same cases, but using a tanker to deliver propellant to LEO.

Without the propellant tanker, the Shuttle-C flight rate may be too high for mixed fleet operations. With the tanker, however, the Shuttle-C flight rate appears reasonable. The propellant tanker flight rate appears slightly high, but these rates are based on a conservative vehicle performance analysis and should go down as more accurate analyses are performed.

The requirements of this case study, when considered alone, do not appear to support the need for a new HLLV, due to the low number of flights for which it would be required. Assuming its only application is the Lunar Evolution case study, the HLLV/propellant tanker combination is also unattractive, since it further reduces the already marginal need for an HLLV.

Mars Evolution Class Mission. The number of ETO flights required to deliver the Mars Evolution LEO facilities, vehicles, payload, and propellant to orbit utilizing the HLLV and the Shuttle-Z are shown in figures 6.7.3-5a and 6.7.3-6a. These figures show only the number of flights required for each Mars launch opportunity, and do not show the actual annual flight rates. The Shuttle-C, due to its relatively limited lift capability as compared to the other launch vehicles, was not considered a viable alternative. Again, propellant is the major component of total mass delivered to LEO.

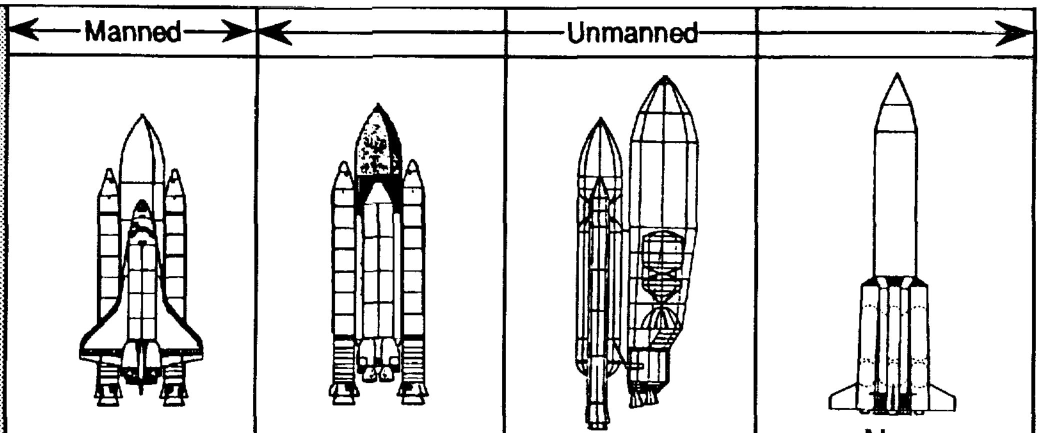
To illustrate the differences in launch vehicle sizing and capabilities, figures 6.7.3-5b and 6.7.3-6b show the effect of accomplishing the same missions with the HLLV and with the Shuttle-Z launch vehicles, with the addition of a tanker to deliver propellant to LEO.

In the cases with no propellant tanker, the HLLV flight rate appears reasonable. In fact, the launch rate shown is probably close to the rate required to justify development of a vehicle with these capabilities. The Shuttle-Z launch rate appears high, since it must share processing facilities with the Space Shuttle, and high launch rates would preclude this.

In the cases utilizing the propellant tanker, the HLLV use is reduced to a point where development probably would not be justified unless additional missions are identified, or multiple manifesting of payloads becomes feasible. If Shuttle-Z with the tanker is the preferred launch vehicle, the launch rate is in a range amenable to mixed fleet operations with the Shuttle. However, the propellant tanker flight rate requirements are very high in both cases.

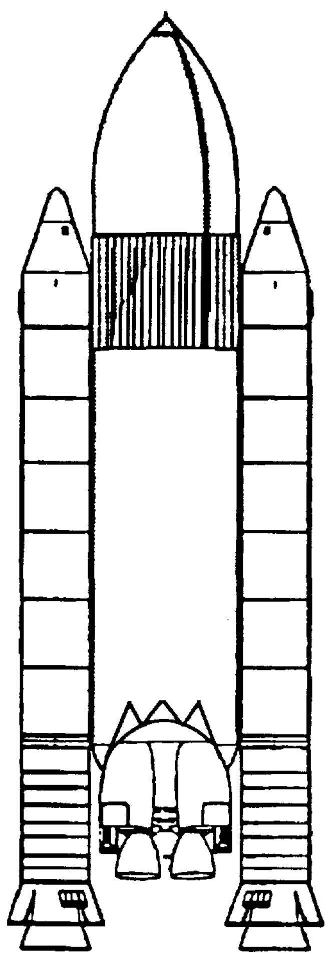
Mars Expedition Class Mission. The number of ETO flights required to deliver the Mars Expedition vehicles, propellant, and payloads to orbit utilizing the HLLV and the Shuttle-Z are shown in figure 6.7.3-7. These figures show only the number of flights required for each Mars launch opportunity, and do not show the actual annual flight rates. Shuttle-C, as currently configured, is not considered a viable alternative. Performance limitations compared to other candidates would result in unrealistically high flight rates (roughly twice the number of HLLV launches). Note again that propellant delivery is the driver for this case study.

The Mars Expedition is the most demanding of the three case studies in terms of launch mass and rate. Although the launch rates, assuming the heavy-lift capability vehicle, are reasonable when compared to today's planned rate capability, the interval over which they are utilized is very short. Development of a new large launch



	Shuttle	Shuttle-C	Shuttle-Z	New Exploration HLLV
IOC	Current	1994	2000	2000
P/L TO 407 km/28.5°	17.7 t at 104%	68.0 t at 100%	87.5 t	156.9 t

Figure 6.7.3-1.- ETO launch vehicle options.



Operational concept:

- Utilize existing Shuttle components.
- Vehicle launches no "payload," but external tank is orbited.

Tanker performance (t)

	RSRM	ASRM	LRB
External tank			
SSME - 100%	74.0	79.5	88.0
SSME - 104%	75.1	80.6	89.0
SSME - 109%	76.2	81.7	89.8
Stretched external tank			
SSME - 100%	78.6	84.0	93.0
SSME - 104%	81.6	87.1	95.9
SSME - 109%	85.3	90.7	99.7

Figure 6.7.3-2.- Shuttle-derived propellant tanker concept.

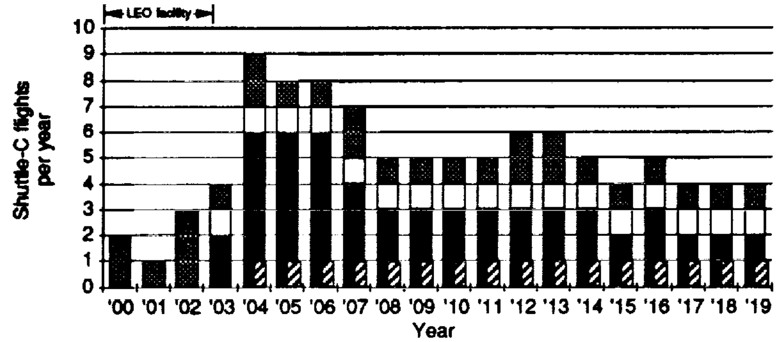


Figure 6.7.3-3a.- Lunar Evolution class mission launch rate requirements for Shuttle-C.

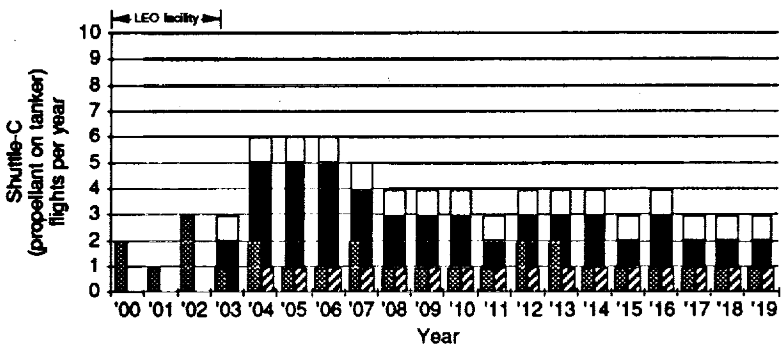
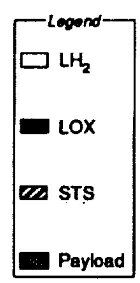


Figure 6.7.3-3b.- Lunar Evolution class mission launch rate requirements for Shuttle-C with propellant on tanker.



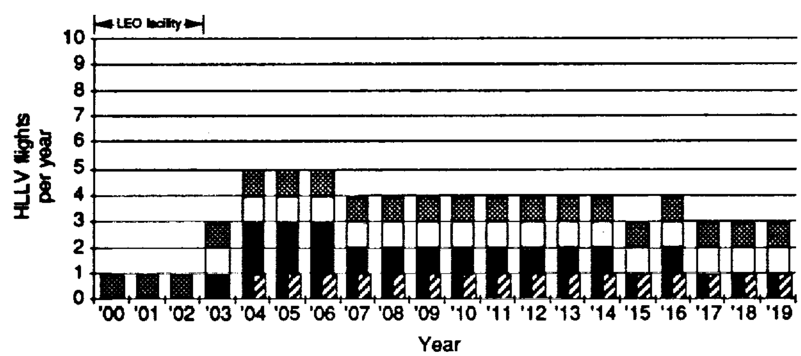


Figure 6.7.3-4a.- Lunar Evolution class mission launch rate requirements for HLLV.

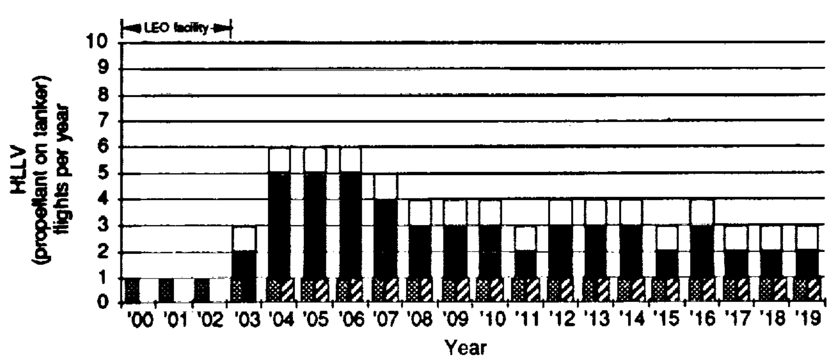


Figure 6.7.3-4b.- Lunar Evolution class mission launch rate requirements for HLLV with propellant on tanker.

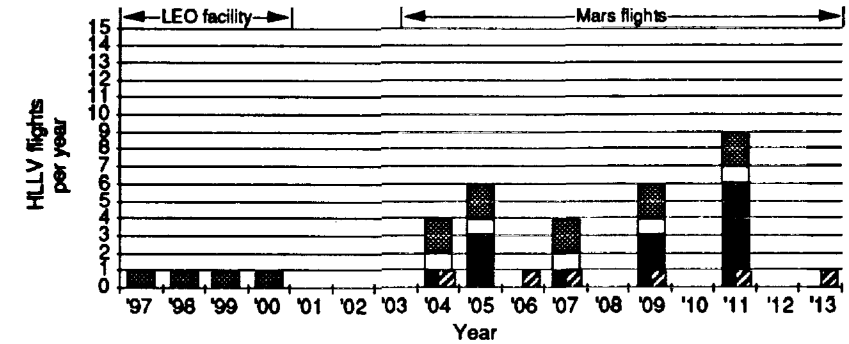


Figure 6.7.3-5a.- Mars Evolution class mission launch requirements for HLLV.

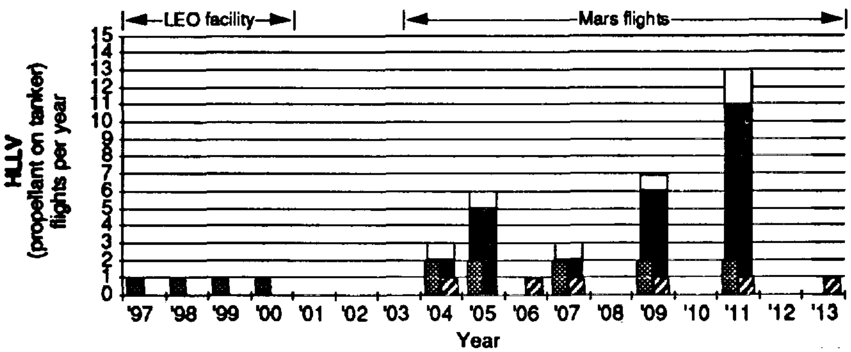
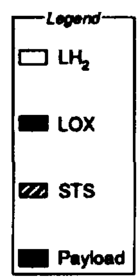


Figure 6.7.3-5b.- Mars Evolution class mission launch requirements for HLLV with propellant on tanker.



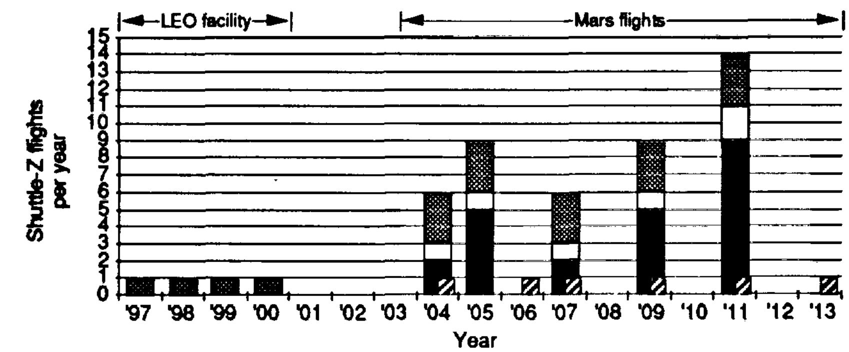


Figure 6.7.3-6a.- Mars Evolution class mission launch requirements for Shuttle-Z.

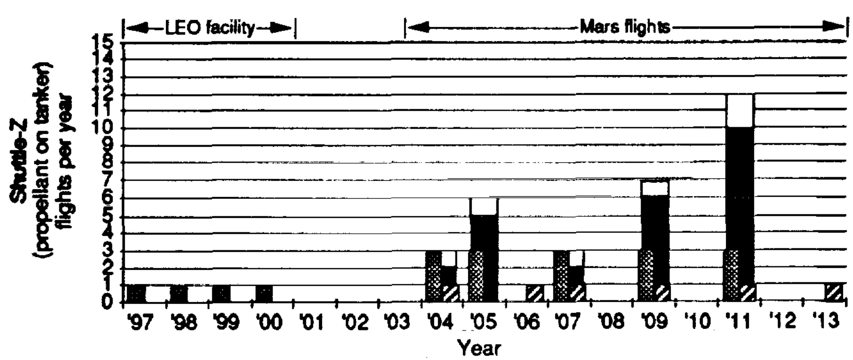


Figure 6.7.3-6b.- Mars Evolution class mission launch requirements for Shuttle-Z with propellant on tanker.

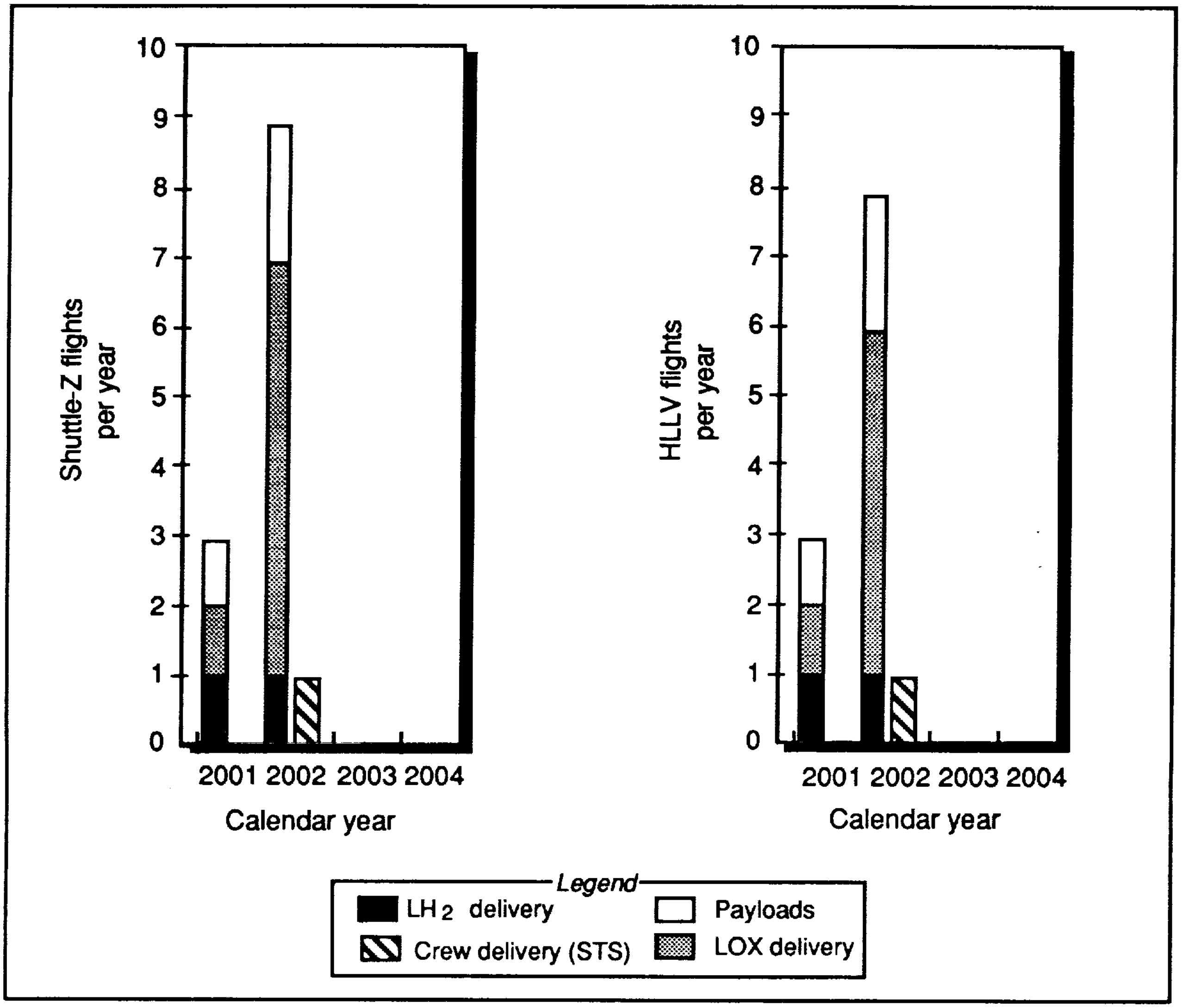


Figure 6.7.3-7.- Mars Expedition class mission ETO requirements.

vehicle or one derived from existing hardware that would be utilized only to accomplish a small number of launches (as would be the case unless future exploration missions are identified), would probably not be economically desirable since this represents a very large investment that has a limited application.

<u>On-Orbit Operations</u>. In the absence of an available on-orbit operations analysis tool or cost estimating capabilities, this study was directed to the identification and definition of other costs. This includes a full assessment of the ground-based facilities and activities. When the on-orbit space operations costs become available, a cost-effective mix of ground and on-orbit activities can be determined.

Table 6.7.3-I shows facility additions to the baseline Space Station Freedom required to support the lunar and Mars case study scenarios. On-orbit facilities are not required for the Mars Expedition-class mission. No on-orbit propellant storage has been included and no logistics requirements have been included in the manifests. The analysis of Space Station Freedom requirements has been documented in Volume IV of this report, and these elements have been incorporated into the ETO manifest.

Operations is the most difficult element of the study to analyze because of the lack of experience in accomplishing the types of space operations required to perform the case study missions in a low-gravity environment. Consequently, computational tools to analyze on-orbit operational requirements (manpower analysis) are still being developed and should be available in early 1990.

<u>Ground Operations</u>. Facility requirements to support the three launch vehicle concepts are shown in table 6.7.3-II. Facilities have been sized to satisfy the most demanding launch rate requirements of the three case studies. Timeline analyses, as illustrated in figures 6.7.3-8 and 6.7.3-9, show the projected processing requirements for

TABLE 6.7.3-I.- SPACE STATION FREEDOM REQUIREMENTS TO SUPPORT EXPLORATION MISSIONS

	Lunar Evolution	Mars Evolution
Habitat	2	1
IVA servicing laboratory	1	—
Resource node	6	1
Cupola	4	—
Logistics module	1	—
Truss bays	175	64
Utility trays	111	64
Solar dynamic units	4	4
Servicing facility	1	1
Airlock	1	—
Mobile servicing center	1	—
Attached payload accommodation equipment	4	4
Lunar STV processing facility	N/A	—
STV orbital support equipment	N/A	2
Thermal radiators	2	1
Docking mast	1	1
Life sciences laboratory module	—	1
CELSS pocket laboratory	—	1
Artificial-g pocket laboratory	—	1

TABLE 6.7.3-II.- FACILITY REQUIREMENTS SUMMARY

Case study \ Mixed fleet	Shuttle + Shuttle-C	Shuttle + Shuttle-Z	Shuttle + HLLV
Lunar Evolution class mission	1 Pad (Shuttle type) 2 MLPs 1 VAB integration bay 1 RPSF 1 CEPF 1 ET HPF (6 cells)	1 Pad (Shuttle type) 1 MLP 1 VAB integration bay 1 RPSF 1 CEPF 1 ET HPF (6 cells)	2 Pads (ALS type) 2 MLPs 1 VIB integration bay 1 CAB 1 LRB HPF (8 cells) 1 LCC
Mars Evolution class mission	1 Pad (Shuttle type) 4 MLPs 2 VAB integration bays 1 RPSF 2 CEPFs 1 ET HPF (8 cells)	1 Pad (Shuttle type) 2 MLPs 1 VAB integration bay 1 RPSF 2 CEPFs 1 ET HPF (6 cells)	2 Pads (ALS type) 2 MLPs 2 VIB integration bays 1 CAB 1 LRB HPF (8 cells) 1 LCC
Mars Expedition class mission	1 Pad (Shuttle type) 3 MLPs 2 VAB integration bays 1 RPSF 1 CEPF 1 ET HPF (8 cells)	1 Pad (Shuttle type) 2 MLPs 1 VAB integration bay 1 RPSF 2 CEPFs 1 ET HPF (6 cells)	2 Pads (ALS type) 2 MLPs 2 VIB integration bays 1 CAB 1 LRB HPF (8 cells) 1 LCC

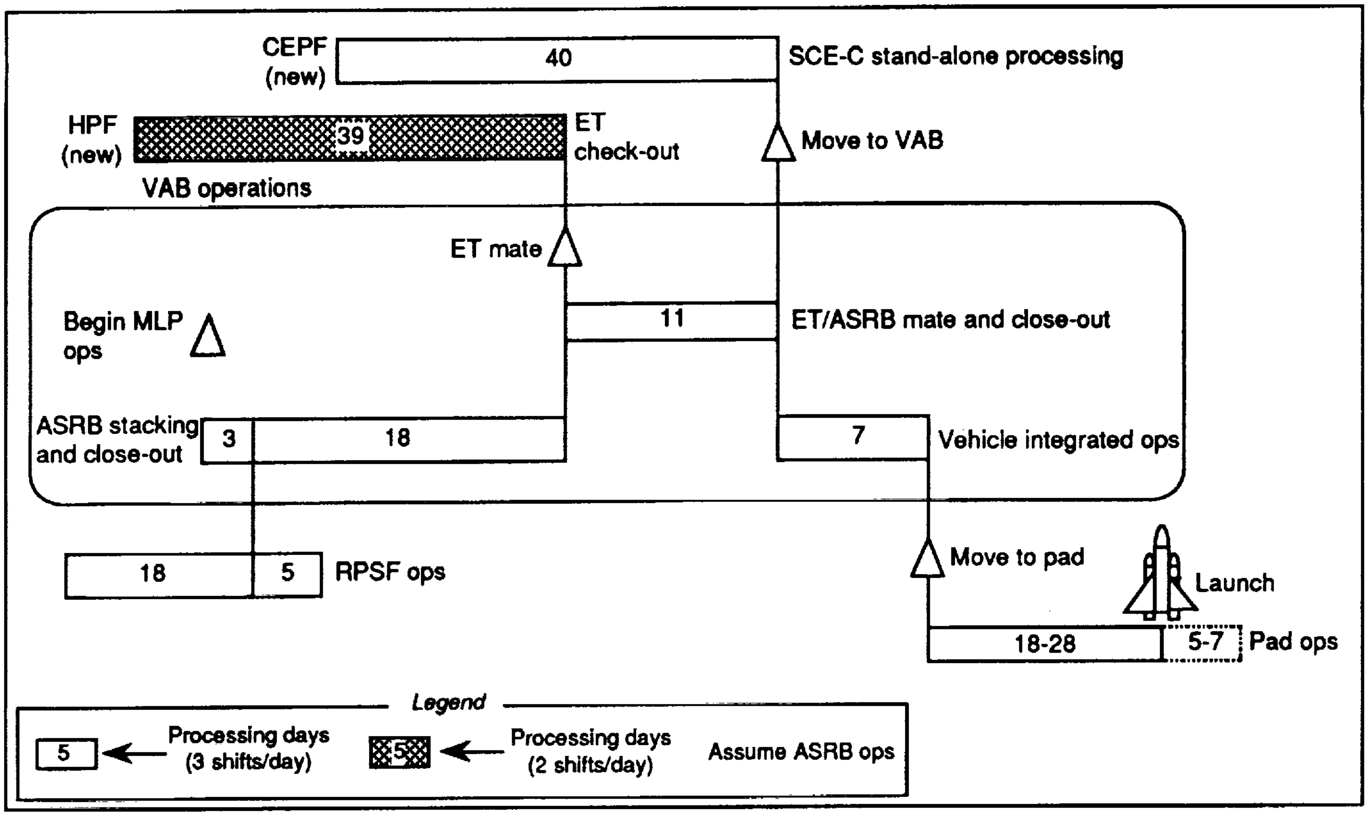


Figure 6.7.3-8.- Shuttle-C/Shuttle-Z processing timeline.

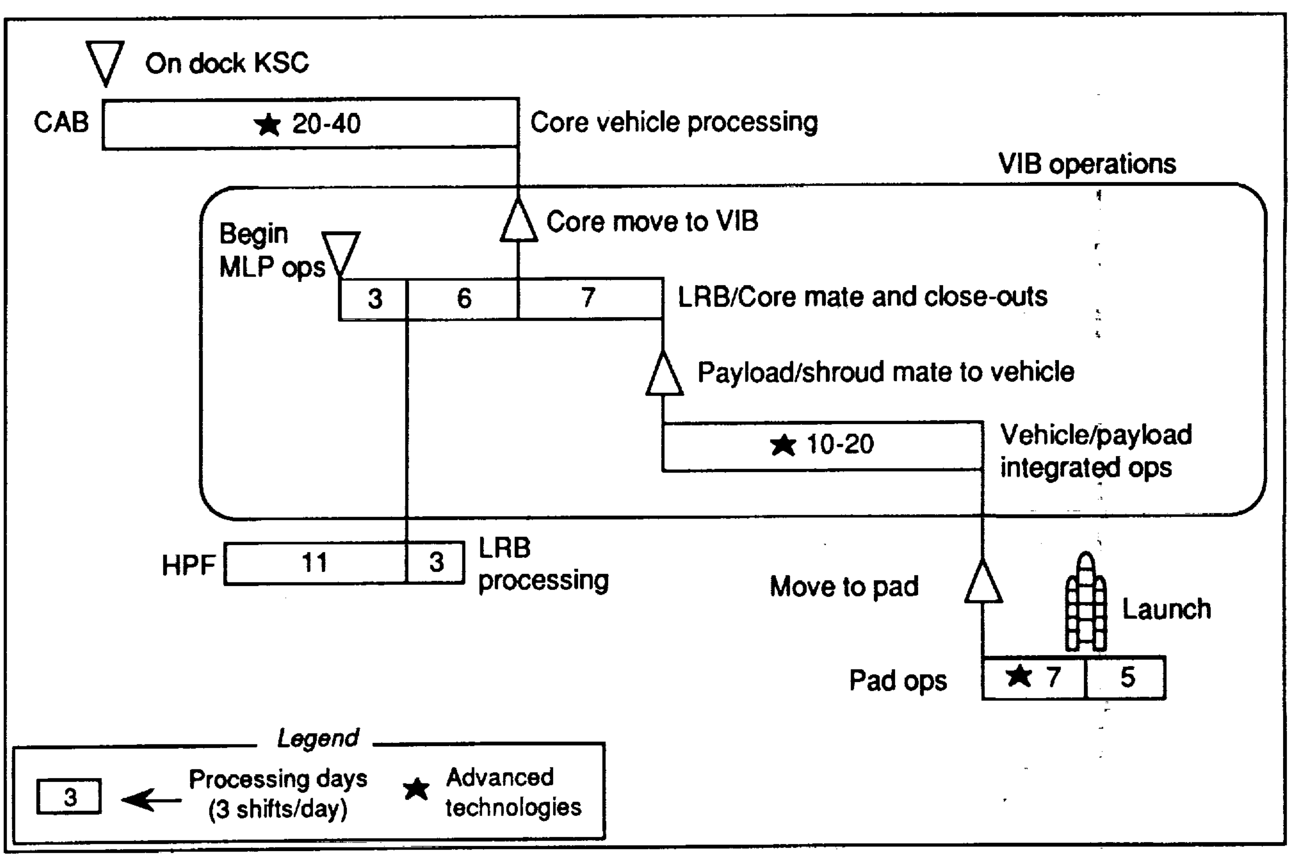


Figure 6.7.3-9.- HLLV processing timeline.

the three vehicles considered. Facility requirements for the propellant tanker concept have not yet been identified.

Facilities for new launch vehicles were based on the assumption that advanced technologies such as automated test and checkout would be in place. This assumption includes all lessons learned from the Shuttle and also includes concepts to minimize vehicle processing critical paths. The Shuttle-derived vehicles were assumed to have improved technologies that include the modifications without adversely impacting ongoing operations.

The ETO facility buildup schedule to meet the requirements of the case studies is shown in figures 6.7.3-10, 6.7.3-11, and 6.7.3-12. The projected relative costs for building these facilities required to support the launch vehicle concepts are shown in figures 6.7.3-13, 6.7.3-14, and 6.7.3-15. STV ground processing facilities are treated separately, and the requirements for these facilities are shown in table 6.7.3-III. For further definition of these facilities, see section 4.1 of this volume.

None of the candidate launch vehicles can be accommodated without significant launch site design, development, test, and evaluation costs. The Shuttle-derived vehicle facilities will have limited growth potential. In addition, the Mars Expedition case study imposes the greatest launch site schedule risk with no schedule flexibility available.

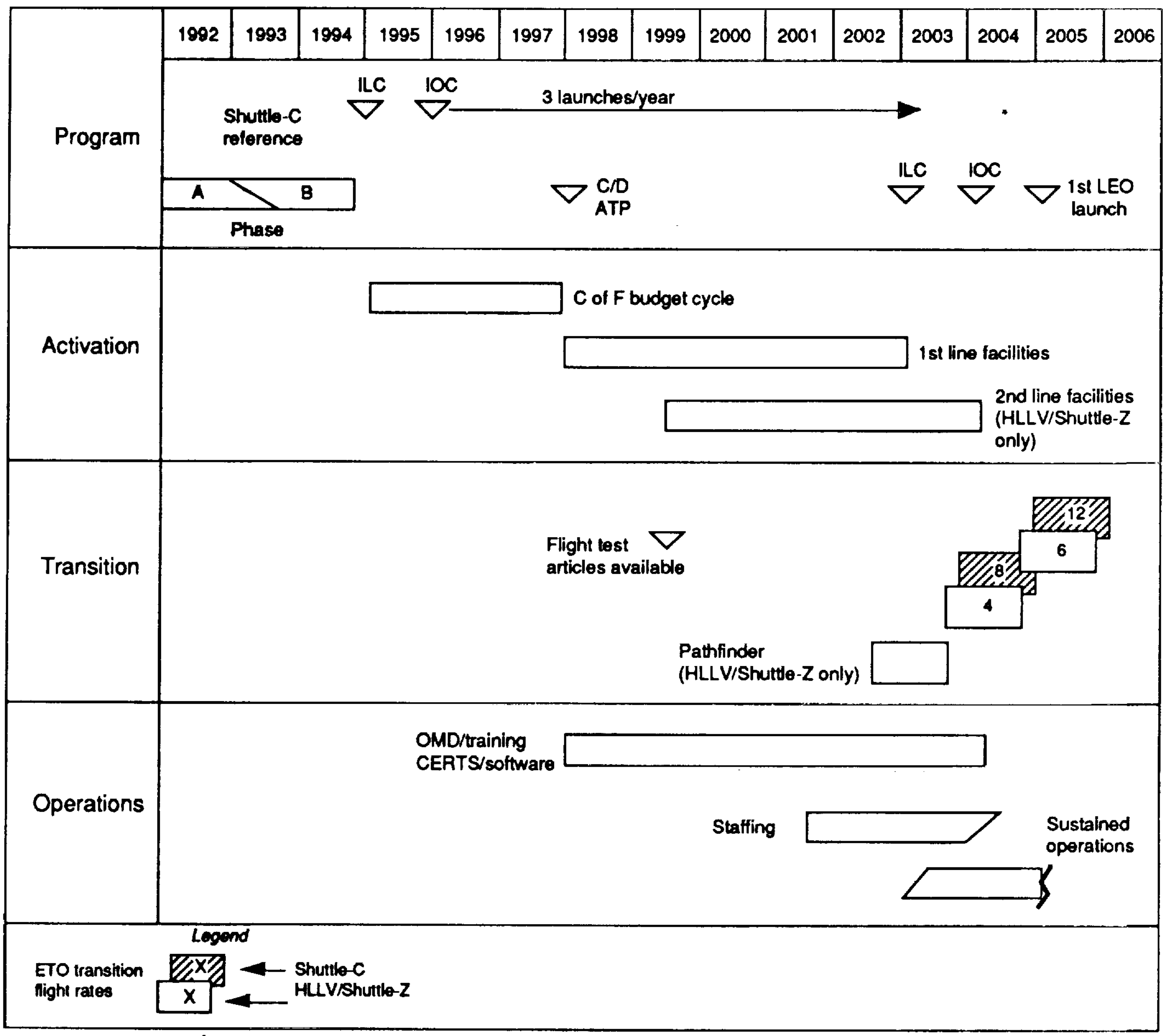


Figure 6.7.3-10.- Lunar Evolution class mission launch site plan.

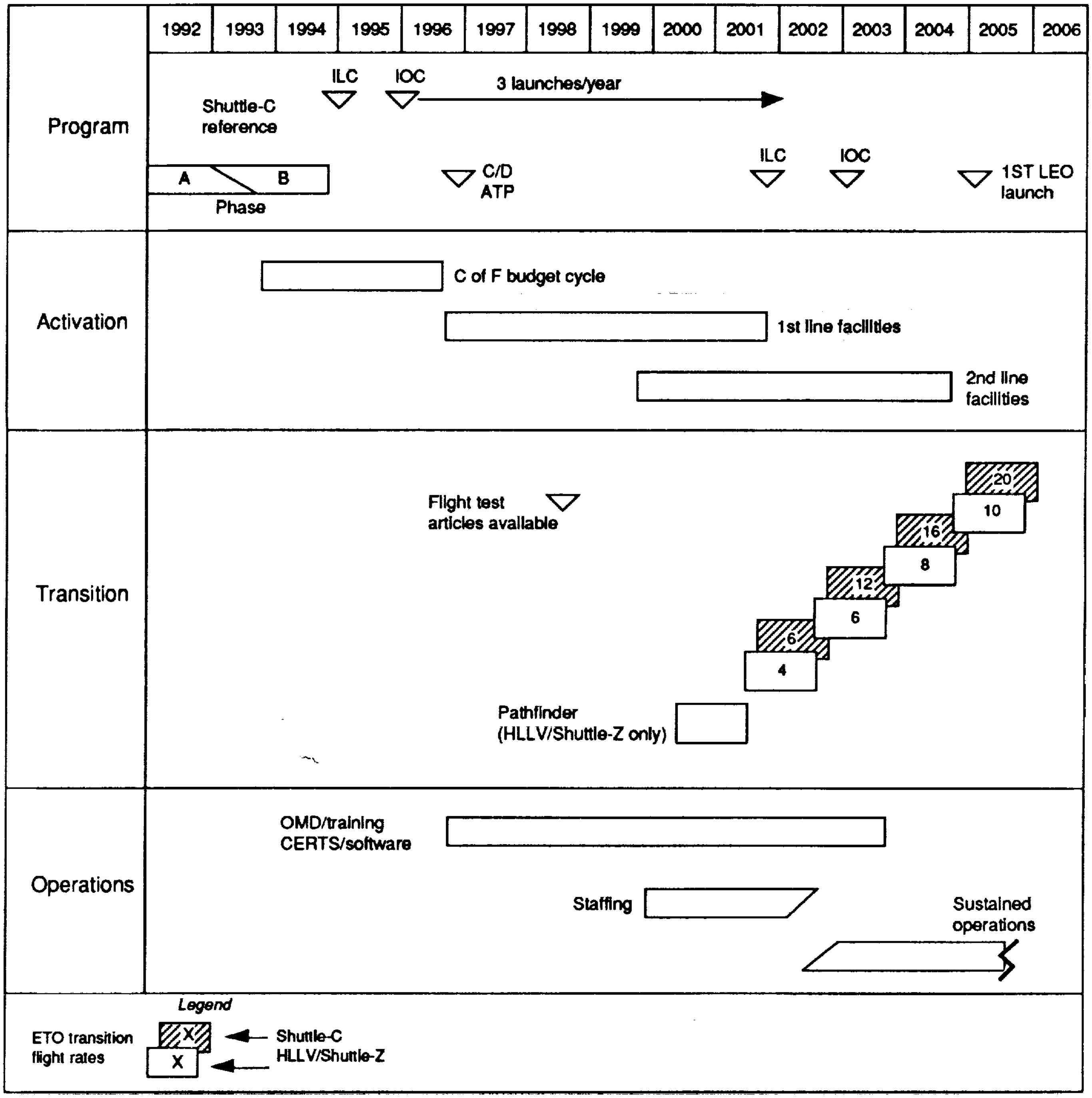


Figure 6.7.3-11.- Mars Evolution class mission launch site plan.

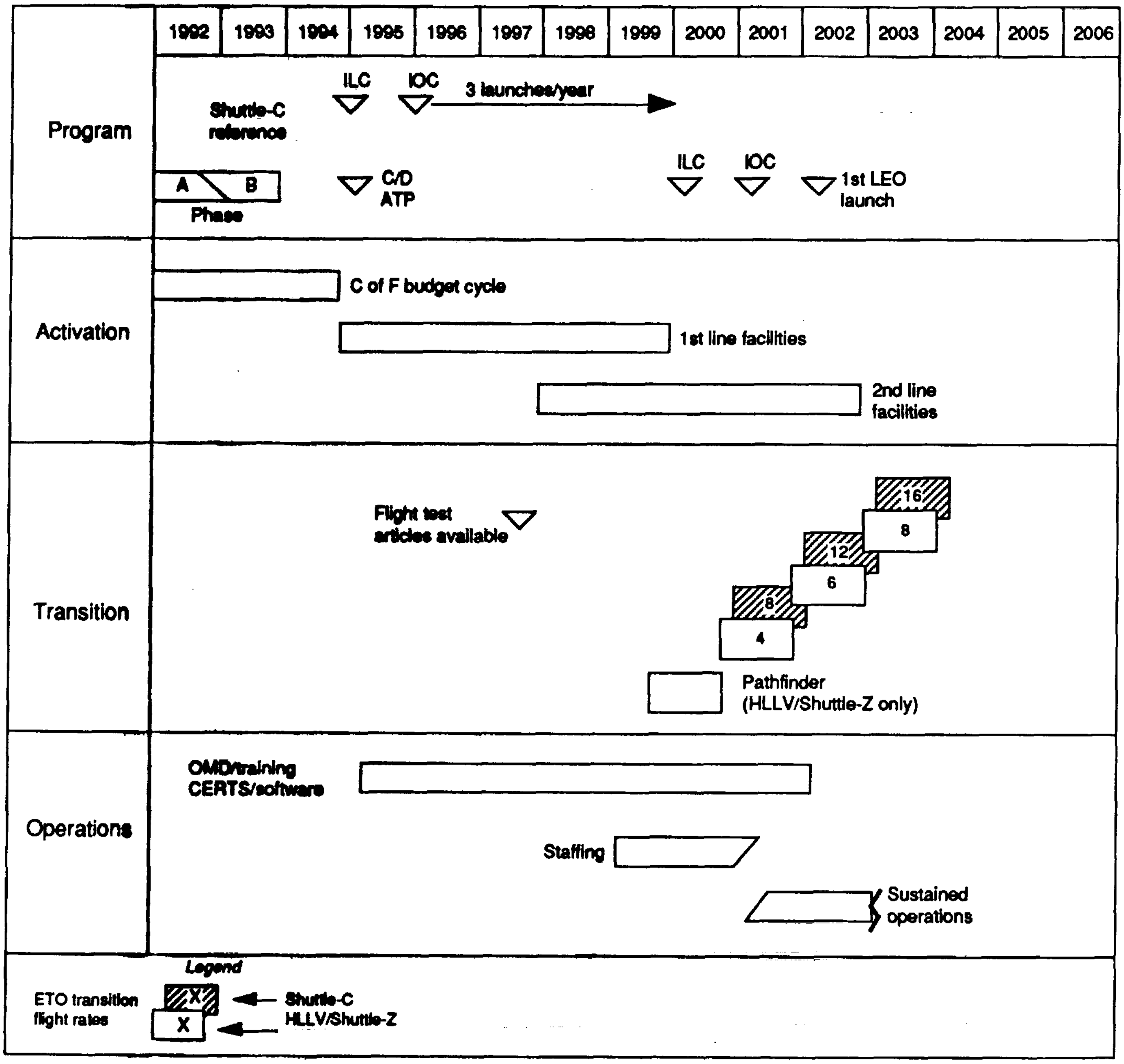


Figure 6.7.3-12.- Mars Expedition class mission launch site plan.

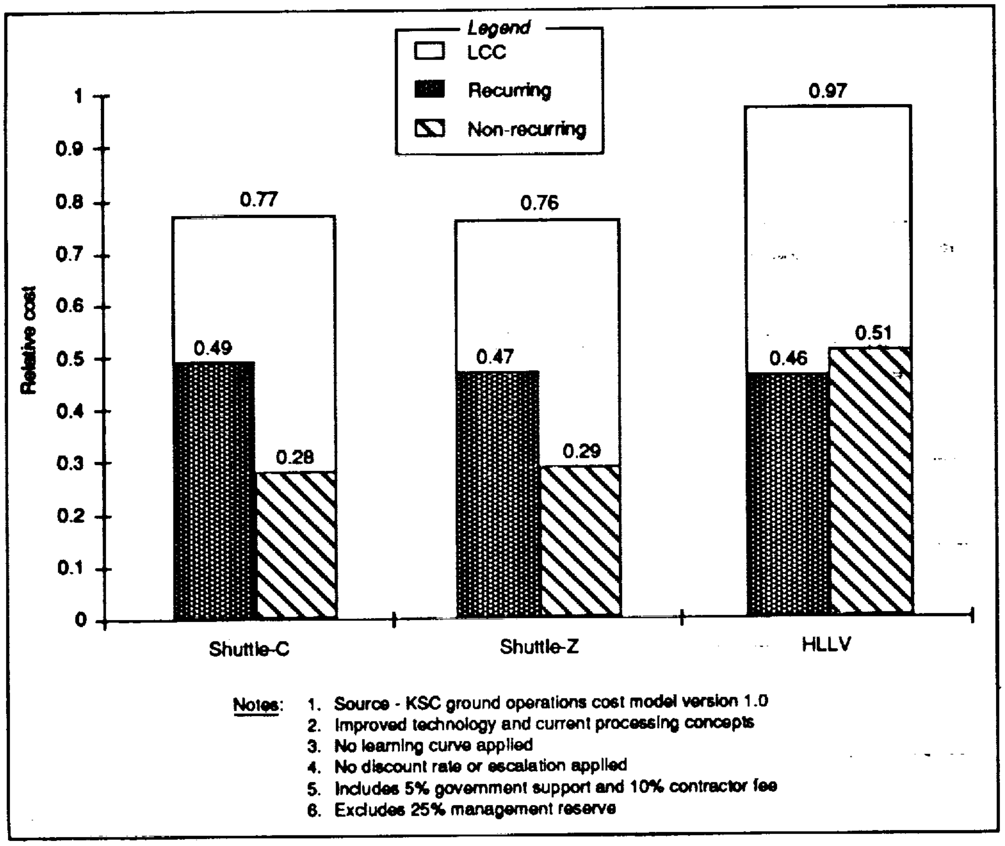


Figure 6.7.3-13.- Lunar Evolution class launch site investment.

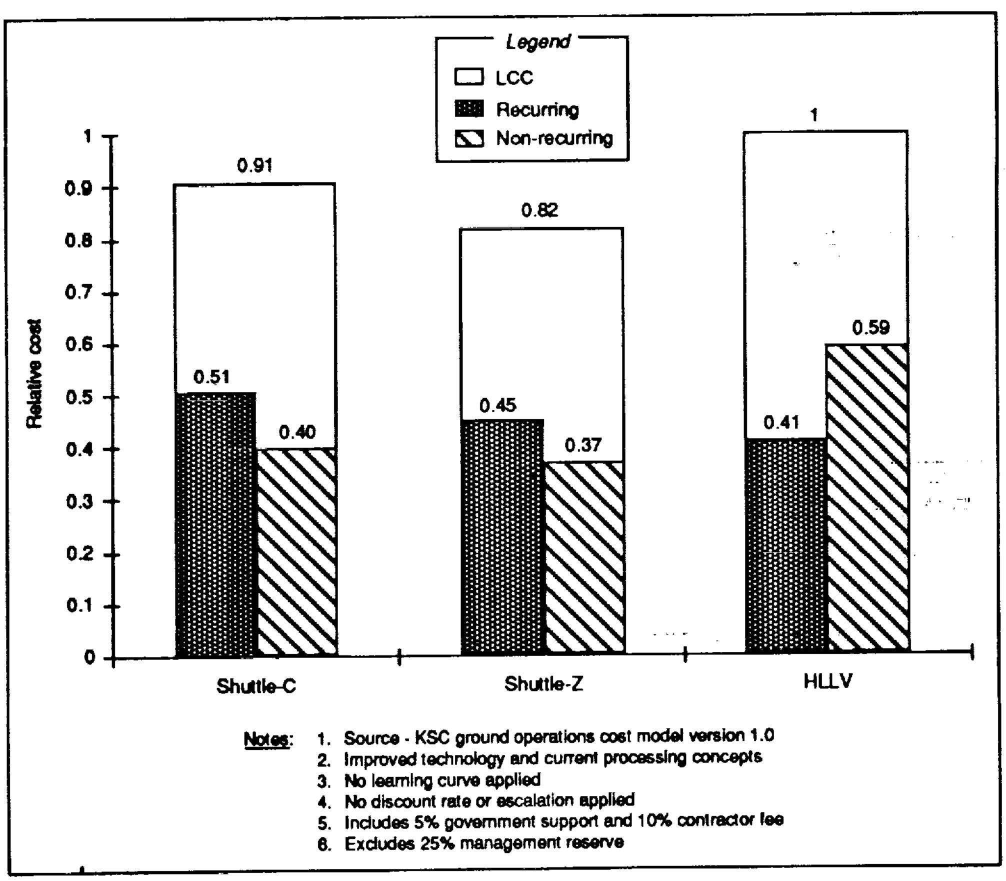


Figure 6.7.3-14.- Mars Evolution class launch site investment.

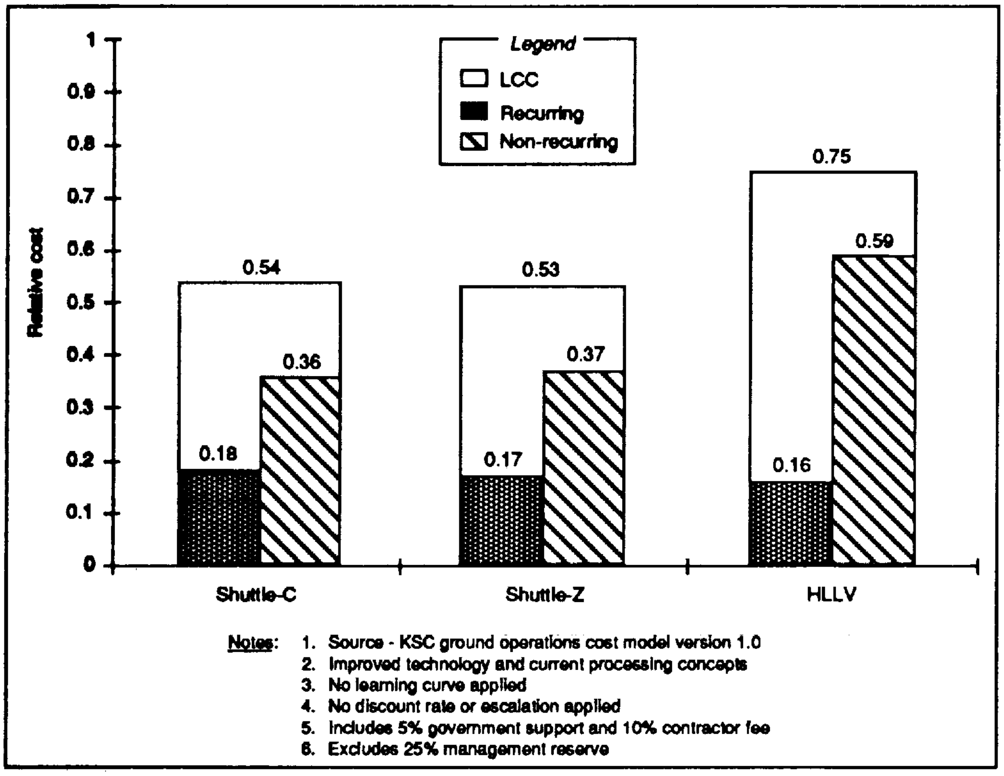


Figure 6.7.3-15.- Mars Expedition class launch site investment.

TABLE 6.7.3-III.- KSC FACILITY UTILIZATION

Facility	Past utilization	Present utilization	Future utilization	Ultimate status
S	DoD-USN	CFES	STS 60, 63, CFES	Possible utilization for RCS and MOOS Processing (non-hazardous)
SSPF	No funding	No funding	Space Station Freedom, Shuttle-C payload processing (no hazardous capability)	Saturation- Space Station Freedom logistics and Shuttle-C payloads
CHPF	Unapproved	Unapproved	Three CERV vehicles in 1995-96 time frame	Could support missions if number of CERV does not increase
IHPPF	Unapproved	Unapproved	OMV	Second "most likely" facility to support missions

Table 6.7.3-IV shows the manpower required to support the peak launch rate as identified previously. Manpower can be identified as a function of yearly launch rate from which the recurring cost can be derived.

TABLE 6.7.3-IV.- LABOR REQUIREMENTS TO SUPPORT LAUNCH VEHICLE PROCESSING

Case study \ Mixed fleet	Shuttle + Shuttle-C	Shuttle + Shuttle-Z	Shuttle + HLLV
Lunar Evolution class mission	8,638 + 4,396	8,638 + 4,183	8,638 + 4,087
Mars Evolution class mission	8,638 + 5,477	8,638 + 4,510	8,638 + 4,087
Mars Expedition class mission	8,638 + 5,326	8,638 + 4,510	8,638 + 4,087

Notes:
1. Source - KSC ground operations cost model version 1.0
2. Operations labor (recurring) only
3. STS labor requirements support an annual flight rate of 14
4. Shuttle-C / Shuttle-Z / HLLV labor requirements support the maximum annual launch/on-orbit flight rates

Legend: Labor given in Personnel Year Equivalents

6.7.4 Conclusions and Recommendations

This study effort has developed the methodology to parametrically analyze and drive out issues associated with the options available for launching and processing STVs. The major limitation is the ability to assess on-orbit processing requirements in terms of operations and manpower. Facility requirements have been identified, but additional study is needed to better define the most efficient division between operations performed on the ground and those performed in orbit.

In summary, during this year the launch/on-orbit processing study defined methodology, developed appropriate analytical tools, and provided initial data for most elements. Some data (especially on-orbit operations) have not been fully developed, and the mission scenarios and vehicle descriptions are continuing to evolve. Further analyses are required to provide support in assessing the impacts of these changes.

6.8 LUNAR OASIS EMERGING CASE STUDY

The Lunar Oasis emerging case study is an effort to define an alternative lunar outpost development strategy that considers self-sufficiency the major goal of the facility. This section defines the rationale of the case study and outlines a reference architecture for the Lunar Oasis in order to determine the magnitude of the space transportation logistics requirements, to understand new technology, and to decide what prerequisite science information should be obtained. With this baseline established, it should be possible to determine the technical and programmatic feasibility of establishing the Oasis. It has been concluded that the Oasis is a feasible approach, and elements of it should be studied in conjunction with future lunar case studies.

6.8.1 Rationale

The Oasis is a concept for a human lunar outpost that is capable of conserving its resources and using indigenous resources to make the outpost nearly self-sufficient, and is eventually capable of expanding with minimum additional material inputs from Earth. The Oasis is used to study human adaptation to the lunar environment in an isolated facility where the crew is resident for years at a time. The crew is involved in scientific experimentation and observation directed in part at the questions of the long-term viability of the outpost, as well as exploring the Moon and using it as a platform for observations of the solar system and universe.

Beginning with the Moon is logical because many of the challenges associated with settling humans on other planets are related to physically and psychologically maintaining people in hazardous and unfamiliar environments. The environments of Mars and the Moon, for example, are both equally inhospitable to human habitation. Although the use of indigenous resources as consumables may be somewhat easier on Mars, the lunar surface is more constant and predictable. The biggest difference between the two bodies is travel time. Since Mars is much farther away, opportunities to rescue crews in cases of system failures, accidents, or sickness are infrequent, and the trip is long. Substantial growth of mankind's technological prowess, understanding of the ability of people to function in isolation, and confidence in the capability of both machines and human beings to succeed in the task will be necessary before a permanent outpost on Mars is feasible. This confidence cannot be gained by sending groups of three or six on sorties into the solar system, but must be gained systematically by establishing facilities in which sizable

crews with a wide range of skills work together for extended periods of time to solve problems associated with residence on another planet.

Attaining self-sufficiency has programmatic and economic benefits. Many current models for lunar outpost programs, recognizing the high cost of transportation to and from an outpost, propose to reduce transportation costs by producing propellants on the Moon, thereby relieving the burden of transporting propellant from Earth into space. Much of the burden, however, is involved with transporting crews to and from the Moon. If facilities can be provided in which the tour of duty of crewmembers can be extended to a period of several years, the burden on the transportation system can be reduced substantially. By reducing the need to resupply the crew with consumables and other imports from Earth, the cost of operating the lunar outpost will also be diminished significantly. After self-sufficiency is established, the continuing operational costs will be relatively low in proportion to the scientific and technological benefits and to other space development and exploration activities that will follow the Oasis.

The Oasis will be partly directed to "operational" scientific research, aimed at understanding the problems of long-term habitation on the Moon. The early science strategy for the Oasis emphasizes using telerobotic explorers outside while the crew analyzes samples and data in lunar laboratory facilities. As experience with the outpost grows, the capability to establish and maintain observatories and laboratories and to conduct long-range surface exploration will grow also.

Human adaptation to a gravity field less than 1-g is not understood. Lunar Oasis will characterize adaptation at 0.16-g, which will be applicable to a Mars outpost (0.38-g). Whereas fractional-g facilities on Space Station Freedom will be quite limited in the number of people who can be tested, data on a larger number of lunar outpost occupants will be able to provide a solid database from which the requirements for Mars missions can be derived.

Although production of propellants for use in the transportation system is not included in the Oasis, the extraction of hydrogen and oxygen from indigenous materials for life support is required. This would provide the technology and experience base for development of a propellant production capability. Materials processing technologies and maintenance and repair facilities would provide the basis for more rapid expansion of scientific capability than if all equipment must be transported from Earth. Development of indigenous power-generating and power-distributing capability could lead in the direction of large lunar power or satellite solar power applications. Thus, the Oasis can enable many alternative future courses of development.

6.8.2 Reference Configuration

Three phases are identified for Oasis development: (1) the oasis phase, (2) the consolidation phase, and (3) the utilization phase. In the oasis phase, the initial facility is established, using a crew of 6 to 10 people. A sortie by a small crew will be made to check out the site and prepare for construction. Equipment delivered in advance will support a 1-year tour of duty for the first permanent crew. The principal components of the outpost at the end of the oasis phase include:

a. The construction facility, which provides temporary space for the crew until the first permanent habitat is erected and activated. Thereafter, the facility serves as a safe haven in case of a catastrophic accident in the habitat.

b. A constructible habitat, the scale of which would be suitable for 10 crewmembers, including all permanent personal and communal space required for the crew, as well as operational capabilities required that cannot be incorporated into the construction facility. The constructible habitat will contain the first phase of the controlled ecological life support system (CELSS).

c. A volatile extraction facility, capable of obtaining nitrogen, hydrogen, carbon, and oxygen and water for the facility by vacuum pyrolysis from the lunar regolith. The volatile constituents are used to make up losses of volatiles from the base, to pressurize constructed facilities, and to fill reservoirs. The volatile extraction facility is essential to the strategy.

d. Science equipment and laboratories, necessary to conduct local geological exploration, conduct remote robotic geological exploration, establish astronomical facilities, and conduct a wide range of research in the life sciences, including research into 0.16-g adaptation, public health aspects of an isolated facility, and investigations of biology associated with the operation of the CELSS.

e. A 1.5 MWt/500 kWe nuclear power station, which will provide electrical energy for the outpost and thermal energy for the extraction of volatiles from the regolith.

f. Surface construction and EVA equipment for construction, maintenance, and exploration activities.

g. A maintenance and repair facility, for analyzing failed equipment and making minor repairs on mechanical and electrical equipment, from a store of component and module spare parts.

At the completion of the oasis phase, the facility can sup-

port its crew with a minimum of resupply from Earth. All volatiles for making up losses from the facility will be replaced by extraction from the lunar regolith, and the CELSS will produce 95 percent of all food required by the crew. At this point, the crew tour of duty can increase to 2 or more years.

At the end of the oasis phase, several courses are possible. The first would be a holding mode, in which the crew tours of duty are lengthened (at a reduction of operational costs), but no additional facilities are emplaced, as the crew reactions are studied and their capabilities in the lunar environment are tested. This course could be consistent with a program that changes emphasis at an early stage to the exploration of Mars. The second course could be a strategy in which propellant production is introduced, in order to reduce the operating costs for a continuing program of lunar development. This is the strategy envisioned by the Lunar Evolution case study described in section 3.2. The third course entails a strategy in which the lunar foothold is expanded and the ability to utilize lunar resources for that expansion is developed.

One approach to the third course is discussed in this section. Additional crew, habitats, and process plants required for internal growth of the outpost are constructed in a "consolidation" phase.

The new facilities to be added during this phase include:

a. Two additional constructible habitats, delivered from Earth, equipped with CELSS capabilities, to increase the outpost staff to 30.

b. A plant for utilizing indigenous lunar resources to produce the materials required for expanding habitable, pressurized volume. For the reference configuration, a concrete-producing system is modelled.

c. A plant for producing metallic iron or iron alloys, which will be used for construction (e.g., as reinforcing in concrete), electrical conductors, and other manufactured products.

d. A capability to produce the elements of a photovoltaic farm utilizing indigenous resources.

e. A repair/maintenance facility upgraded with manufacturing capabilities.

During the consolidation phase, the crew tour of duty will increase to 3 years or longer. At the end of this phase, the outpost will be able to expand its pressurized volume and power output and provide for many of its internal needs utilizing indigenous resources. The last phase is called the "utilization" phase, and it is not described here.

Table 6.8.2-I summarizes the general characteristics of the three phases of the Oasis.

6.8.3 Approach

Over the past few years, descriptions of the major components of the Lunar Oasis reference configuration have been presented among a growing number of concept studies. The Oasis configuration has been assembled from such elements scaled to the requirements of the Oasis crew and their activities. The "mission manifest" that catalogs the elements, their capabilities, and their sequence of emplacement is outlined in table 6.8.3-I. For the purposes of the study, it has been assumed that the space transportation system consists of Earth-to-orbit vehicles, orbital transfer vehicles, and lunar lander and ascent vehicles capable of transporting 20 t to the surface in one-way cargo missions (expendable landers) and 14 t on round-trip piloted missions. The transportation system must deliver three payloads to the Moon each year for 10 years.

6.8.4 Principal Elements of the Oasis

The principal components of the facility are discussed below. References are provided to the literature sources for the concepts. More complete descriptions and scaling assumptions, as well as assumptions of the interdependence of various elements of the outpost, can be found in Volume VI of the Exploration Studies Technical Report.

1. Constructible Habitats: One of the characteristics of the Oasis is that substantial amounts of pressurized volumes are established. A concept for an inflatable structure with a volume of approximately 2,000 cubic meters has been proposed by M.L. Roberts in a 1988 paper, *Inflatable Habitation for a Lunar Base.* The use of such a structure substantially reduces the mass that must be brought from Earth for habitats, laboratories, and the CELSS modules. It is not evident that these will be suitable for the "industrial facilities" considered in the strategy, which may also be constructible. Alternative approaches to constructible structures should be considered, including concepts involving tunneling within the regolith (ground-up rocks and glass covering the lunar surface to a depth of several meters) or into the rock below the regolith. Three habitats and two industrial facilities, with a total usable volume of 10,000 cubic meters, are required.

2. Volatile Production: The requirement for substantial amounts of pressurized habitable volume and a self-sufficient life support system makes the production of gases and water for pressurization and reservoirs essential to the Oasis strategy.

TABLE 6.8.2-I.- CHARACTERISTICS OF LUNAR OASIS PHASES

Subsystem	Oasis phase	Consolidation phase	Utilization phase
Habitat	Constructible habitat from Earth	Add habitats	Lunar concrete for pressurized structure/ other materials for internal use
Volatiles handling	Establish and test systems	Fill consumable reservoirs	Pressurize new volumes
Life support system	CELSS facility ECLSS backup	System activated 95 percent food produced	Augment CELSS
Power	MWe nuclear	Research options for solar power	Expand power using local materials
Resource processing	Mining/ processing for volatile plant	Metals plant	Production of new structures and utilities
Manufacturing	Small machine shop for maintainance and repair	Explore manufacturing of significant outpost elements	Manufacturing supports outpost expansion
Science capability	Supports outpost functions Local exploration	Growth of science labs and observatories	Growth of science capability using indigenous resources

TABLE 6.8.3-I.- MISSION SEQUENCE AND ACTIVITIES

Year	Number	Description	Crew activities
1	1	The construction module is a Space Station Freedom derivative, and has a self-contained life support system	
1	2	Power system, construction equipment, navigation aids, and supplies to the construction module	
1	3	4-person crew/4 months; space suits; provisions; spares	Temporary crew reconnoiters site, activates construction module
2	4	Supplies for a 1-year tour of duty; additional EMU hardware; science equipment (science equipment is carried by many missions)	
2	5	Integrated 0.5 MWe nuclear power facility and volatile extraction facility. Volatile extraction facility provides gas for pressurization of constructible habitat, and to make up for system losses	
2	6	6-person crew/1-year tour of duty; tools; repair/ maintenance system; communications equipment	Assemble, activate nuclear power/ volatile plant. Science stations erected and exploration carried out throughout program
3	7	First constructible habitat	Assemble, pressurize habitat
3	8	CELSS system, suitable for supporting 10-person crew	Assembly and check-out of CELSS system
3	9	10 crew up/6 crew down; pressurized rover; science equipment; provisions; spares	Outpost operations, check-out, and system experience

TABLE 6.8.3-I.- (CONCLUDED)

Year	Number	Description	Crew activities
4	10	Second nuclear power/volatile extraction module	Assemble and activate power/volatile plant
4	11	Supply mission; space suit systems	
4	12	10 crew up/10 down; science equipment; provisions; spares	
5	13	Constructible habitat to provide for 10 additional crew	Erect, pressurize habitat
5	14	CELSS system	Install CELSS
5	15	10 crew up	20 crew Tour of duty is now 2 years (for previous crew)
6	16	Augmentation to volatile facility, doubles output	Install facility
6	17	1 MWe nuclear power	Activate power plant
6	18	10 crew up/10 down; science equipment; provisions; spares	
7	19	Industrial module, to house elements of metal and concrete production facilities	Install module
7	20	Metal manufacturing facility	Establish facility
7	21	10 crew up/10 down; provisions; spares; science equipment	
8	22	Concrete manufacturing facility	Install, check out facility
8	23	Constructible habitat, to increase outpost capability to crew of 30	Erect habitat
8	24	10 crew up/10 down; space suit replacements; spares; science equipment	
9	25	1 MWe nuclear power system	Install power system
9	26	CELSS system, to provide for 10 additional crew	Install CELSS
9	27	10 crew up; spares; provisions; space suits and maintenance system	30 crew; 3-year tour of duty
10	28	Industrial module, to provide space for solar cell manufacturing facility	Erect facility
10	29	Solar cell production equipment	Install equipment
10	30	10 crew up/10 down; spares; provisions	
11	31	Solar cell production equipment	Complete installation, initiate production

Note: Every third mission is a piloted mission. Cargo missions carry a payload of 20 t; piloted missions, 14 t.

The Moon is characterized by the absence of oceans and atmosphere, and the presence of rocks that are quite dry. Nevertheless, the lunar regolith contains substantial quantities of volatile elements, implanted over eons by the solar wind, which can be extracted from the regolith by heating at 600 to 900° C. Of the gases needed to pressurize the Oasis, nitrogen is rare in the lunar regolith, where a typical concentration is 50 ppm. Concentrations of hydrogen and carbon are slightly higher. The amount of nitrogen needed for pressurization depends on the selection of atmospheric pressure and composition for the lunar facility. The Oasis model assumes an internal pressure of 0.5 atmosphere with 40 percent oxygen. This requires only 3/8 of the nitrogen that would be required for one atmosphere pressurization.

The principal losses from the facility are from diffusional losses and the operation of airlocks. In the oasis phase, it has been assumed that volumetric diffusion losses are on the order of 6.9 kg/day from the 2,000-cubic meter constructible habitat and that four daily operations of airlocks lose a total of 1.4 kg/day. These assumptions

are scaled to later phases according to the number of pressurized structures. Larger loss rates will more strongly require the generation of nitrogen from the lunar regolith. Smaller rates will reduce requirements for power or allow other uses for the volatiles.

A facility for volatile extraction was described in a 1988 Eagle Engineering Inc. study of a lunar oxygen pilot plant. In order to produce the amount of nitrogen required in the oasis phase, scaling that facility indicates that a system mass of 9,700 kg, 800 kWt, and 300 kWe will be required. Much of the heat could be supplied by waste heat from the nuclear power system.

Hydrogen is produced in the volatile production facility in about the same quantities as nitrogen. Since hydrogen is not required for habitat atmosphere, and it is difficult to store, it is assumed that the hydrogen is reacted with oxygen to form water to be used elsewhere in the system. Similarly, it is assumed that the carbon released from the soil will appear as carbon dioxide and will be stored for eventual use in expanding the life support system. During the utilization phase, water is necessary for use in lunar concrete construction and strongly influences the volatile production requirements.

The total annual yield of useful volatile products from the facility is estimated to be approximately the same mass as that of the machinery needed to extract it (without considering the power supply). Recent analysis by E. Christiansen (personal communication) suggests an even shorter mass recovery period.

In the process of producing water and carbon dioxide, metallic iron will be formed. If it is assumed that the process is efficient, sufficient metal is produced to meet the construction needs of the facility in the utilization phase.

Although solar wind gases are present in the regolith, the content is a function of the "maturity" of the lunar regolith. Thus, areas of higher maturity will be more favorable for the Oasis, other considerations being equal. For example, the difference between a site with 50 ppm nitrogen (assumed here) and 75 ppm nitrogen (known from samples of lunar soil) would be a 1/3 reduction in the size of the volatile extraction facility and its associated power plant. It may be possible to determine the maturity of lunar regolith by remote methods as described in M.P. Charette's 1976 paper *Age-Color Relationships in the Lunar Highlands*. The appropriate sensors are under consideration for the Lunar Observer mission now being planned by NASA. Such a mission could be effective in selecting an optimal site for the Oasis.

3. Controlled Ecological Life Support System (CELSS): The CELSS is also critical to the success of the Oasis. It is assumed that 95 percent of all food will be provided by the CELSS, which also recycles all air and water. Because the air and water reservoirs grow with time as a result of volatile production, it is possible to store a portion of the waste products for extensive periods of time, for example, by utilizing slow biological waste treatment methods. It is assumed that the provision of extensive reservoirs, the existence of multiple habitats, and the reliability of a modular CELSS system will make the provision of physical/chemical life support systems unnecessary beyond the initial construction facility. Nevertheless, physical/chemical waste reactors may be incorporated into the CELSS. CELSS systems designed to date generally have been assumed to be hydroponic. Because of the availability of lunar surface materials, soil substrates may be desirable. Such substrates could allow easier expandability in the utilization phase of the Oasis. The mass of the CELSS required for support of four persons is estimated to be 10 t with a power requirement of 200 KWe.

4. Power Systems: The initial construction facility is assumed to be a modified Space Station Freedom module, with self-contained life support and thermal control systems, which provides living space for the construction crew and is used subsequently as a work area or safe haven. A 25-kWe photovoltaic array with regenerative fuel cell storage device provides power to the construction facility. The mass of the self-contained habitat is 20 t. The 25-kWe power supply has an additional mass of 10 t.

For the Oasis, an SP-100 derivative nuclear reactor with 1,500 kWt capability with eight Stirling engines capable of generating an average of 500 kWe has been assumed. This power plant is scaled from the system described by L.S. Mason and H. Bloomfield in a 1989 paper entitled *SP-100 Power System Conceptual Design for Lunar Base Applications*. For safety, the reactor is buried in the regolith and has independent heat rejection loops for each engine. The SP-100 nuclear reactor design weighs 10 t and has an expected lifetime on the order of 7 years. Thus, in later stages of the strategy, a replacement reactor will be required. Perhaps the first reactor could be reconfigured to provide only thermal power and remain useful for a longer time.

5. Concrete Production Facility: One of the concepts discussed recently is that the material required to produce concrete can be obtained from indigenous lunar materials and that freestanding structures will be possible using concrete as a construction material. A volatile production facility provides the water required for the facility. The lunar concrete building described in a 1988 paper by T.D. Lin et al was scaled to 2,000 cubic meters, equivalent to that of the constructible habitat. Concrete production was assumed to require the sepa-

ration of rocks, pebbles, and sand-sized material from the regolith for use as aggregate, production of cement by the chemical processing of lunar plagioclase feldspar ($CaAl_2Si_2O_8$) and production of water. Metal reinforcing and tension cables are produced by the metal facility described below. In order to produce 2,000 cubic meters of habitable volume per year, the mass of the facility is estimated to be 15.5 t and to require 200 kW of electric power. This does not include the requirements for iron products. It does include all mining, beneficiation, chemical processing, mixing, casting, curing (including an autoclave for rapid curing of precast concrete slabs), and surface transportation systems for an end-to-end concrete construction capability.

6. Metals Production: Metallic iron occurs naturally in the lunar regolith in small quantities. In addition, any process that extracts oxygen from lunar silicates or oxides produces iron as a by-product. For this study, it was assumed that metal is the by-product of oxygen production and can be extracted at elevated temperatures from that process. A very simple system for melting the concentrated iron, casting it into ingots, and rolling the ingots into bars and rods, pulling wire, and manufacturing wire cables was sized to produce 150 t (about 20 cubic meters) of iron per year. The system was based on data from *The Making, Shaping, and Treating of Steel*, a 1971 publication by the United States Steel Corporation. This system is about twice the throughput that would be required for the concrete building fabrication. A preliminary mass of 10 t is estimated with a power requirement of 100 kWe.

7. Solar Photovoltaic Device Production: The objective of the facility would be to produce photovoltaic devices and structural materials on the Moon. The system is scaled from a General Dynamics Convair study that defined a lunar and space facility for producing silicon photovoltaic cells, fused silica cover glasses, and supporting structures for satellite solar power systems. The facilities defined in that study were scaled to the production facility required for 10 MWe of new power per year on the Moon. A target of no more than 10 t of terrestrial materials per each 10 MWe lunar photovoltaic array would be a significant performance improvement over nuclear plants brought from Earth.

Silica glass and silicon metal would be produced in association with the processing of feldspar for cement. One of the products obtained from the decomposition of feldspar is silica, and the amounts obtained are approximately equal to the amount of cement produced. This amount is approximately 10 times the total needed for the silicon solar cell production.

Estimates of the mass and power of equipment necessary to produce silicon solar cells at a lunar facility are 30 t of equipment and 365 kWe. The scaling from the General Dynamics Convair study includes a factor that reflects the 50 percent duty cycle of the lunar facility compared to the free space processing facility envisioned there.

8. Science Facilities: The science equipment for the Oasis reference configuration will not all be delivered at one time, but instead is included in many of the manifested flights. Some of the science equipment will be cross-utilized for operational support, in particular the analytical facilities, which will perform monitoring services and study failure problems as well as performing fundamental scientific investigations. Some operational facilities will include science capability. These facilities include the Health Maintenance Facility, which will be used for studies of crew health and fitness, and the CELSS, which will include monitoring and analysis capability in support of that facility. Other facilities to be included are:

a. Geological field tools and emplaced experiments

b. Petrological analysis laboratory

c. Biological analysis laboratory

d. Animal holding and experiment facility

e. Plant experiment facility

f. Astrophysical, solar and terrestrial observatory

g. Experimental petrology/materials processing laboratory

h. Repair and maintenance facility.

6.8.5 Discussion

Scale of Program. The delivery of components up to the completion of the consolidation phase requires 31 missions over 10 years. It is assumed that these missions can be carried out at 4-month intervals, or at the rate of three per year. The magnitude of the entire scenario is a little more than four times that of the Apollo Program, as each of these missions is of similar magnitude to an Apollo flight. The ratio of cargo to personnel flights is 2:1. This distinguishes the Oasis from the Apollo Program, in which no permanent support equipment was delivered to the Moon. The strategy is designed to limit the number of crew missions to one per year. For example, as the crew increases from 10 to 20, it is assumed that the tour of duty increases from 1 to 2 years.

At the end of the consolidation phase, the capability exists to expand habitable volume, life support, and power largely using the Moon's indigenous materials. In contrast to the Apollo Program, which was exploratory and self-contained, leaving no installed facilities at

the end of the program, the Oasis is capable of nearly unaided expansion. At the end of the consolidation phase it could survive for long periods of time with no resupply from Earth.

Integration of Production Facilities. A variety of interrelationships exists between production facilities and the power systems. The temperature of the nuclear reactor is slightly higher than that needed to extract volatiles from the regolith, and the rejected heat could be useful as preheat.

The by-products of volatile extraction can become the feed to the metals-producing facility. The production of cement can provide silica as a by-product for solar cell production. An integrated power station/volatile extraction/materials production facility should be considered, although it will present significant design challenges.

Self-Sufficient Growth. The elements that allow the facility to grow after the consolidation phase include concrete and metal facilities, which allow new pressurizable structures to be constructed using a minimum of materials brought from Earth. In early stages of expansion, it is envisioned that interconnecting airlock modules and bladders for containing pressure may have to be provided from Earth; however, all other construction materials can be lunar. Growth includes expanded volatile production allowing the pressurization of new habitats and the production of photovoltaic power systems, which include silicon photovoltaic devices, supporting structure, and appropriate electrical connections. All, in principle, can be fabricated from lunar materials, with minor quantities of materials (e.g., precious metals) brought from Earth. In addition, capabilities will exist for constructing an expanding network of roadways and unpressurized structures.

The concrete and metals production facilities in the Oasis have been sized to create the equivalent of one pressurized habitat per year. There is excess capacity in both systems at that rate, and other structures or uses would be possible. The initial sizing of the photovoltaic system production facility is for 10 MWe per year. This also is very large compared to the local requirements. However, rapid increase in the availability of power will open many new options to the Oasis.

No provision has been made for adding staff or facilities beyond the start of the utilization phase. It is assumed that the requirements for additional people and equipment will be developed by the time the utilization phase starts.

Technology Requirements. A number of technologies must be demonstrated if the Oasis is to be successful. Some of the more important include:

a. MWe nuclear power generation

b. Low-mass constructible habitats (e.g., inflatables)

c. Controlled ecological life support systems (95 percent + closure)

d. Lunar materials processing (mining, extraction, and bulk product preparation) for:

   (1) Volatile extraction

   (2) Cement production and concrete preparation and handling

   (3) Metal production

e. Computer-controlled flexible repair and manufacturing

f. Photovoltaic devices, structures, and energy storage and distribution utilizing indigenous materials

g. Improved performance of analytical and experimental laboratory instruments

h. Automated surface rovers

i. Large pressurized facility design and construction

j. Hermetic sealing of facilities constructed with indigenous materials.

There are a number of areas in which competitive approaches to the reference concept should be developed, including subsurface tunneling as an alternative to constructible habitats, and sintered regolith as a construction material.

Crew Considerations. Many additional studies are suggested by this analysis. One interesting set of questions deals with the capabilities of the crews that will occupy the Oasis. The objectives of the Oasis are to demonstrate self-sufficiency, to develop confidence in maintaining crews for periods of years at a planetary surface outpost, and to conduct scientific experimentation and observation. The performance of the crew will be key to the success of the Oasis. Because this will be a novel undertaking, and one in which people on-site should have a major input to the development of capability, new levels of autonomous action by the outpost members will be sought.

The tasks of installing, maintaining, and operating Oasis systems and conducting research are widely diverse, requiring significant amounts of pre-mission training, cross-training, and individual flexibility. Other considerations in selecting crewmembers will include potential for innovation, technical capabilities compatible with outpost operational modes (e.g., computer/automation), and personal characteristics (capability to work in a team). This should be an area of intense investigation

as preparations are made for the Oasis.

An initial examination of crew skills and jobs was made. Because of the breadth of skills required, it is clear that all outpost personnel will have to have multiple duties, either cross-disciplinary or combinations of management and technical tasks. Some tasks, such as facility housekeeping, will probably be shared equally by all crewmembers. Even a cursory consideration of this problem indicates that true autonomy by the crew, in terms of being able to function without technical support from Earth, is unlikely.

Two areas deserve special attention. The first is the relationship of people and machines. Most of the routine work at the Oasis must be done by machines. Humans will have the role of directing the efforts of the machines and keeping them in operating condition. The people should also be keen observers of machine capabilities and potential, so that improved versions of hardware and software can be developed.

Another key consideration is ground simulation. It would appear reasonable to operate a virtual replica of the Oasis on Earth, where personnel on the ground can understand the long-term capabilities of both machines and crews and can serve as problem-solving aids to the lunar crew. If new applications for lunar materials, machines, or people are identified by the Oasis crew, it should be possible to rapidly develop them on the ground and communicate solutions to the crew. The design engineering staff responsible for the Oasis should have access to this facility. Provision for the ground facility and supporting staff should be included in programmatic plans for the Oasis.

<u>*Precursor Science Missions and Program*</u>. Among the Apollo sites, any mare site would appear to be a reasonable choice from the point of view of volatile extraction and system performance. The Apollo 17 site, where anorthite- and ilmenite-rich rocks and volcanic glasses with surface concentrations of metals and volatiles were found, would provide a suitable location. However, an orbital survey for regions of mature regolith is a highly desirable precursor to the Oasis. Discovery of areas of high volatile content can improve system performance significantly.

The scientific objectives of the Oasis will be enhanced by further understanding the Moon through global mapping such as that proposed for the Lunar Observer. Because a major activity at the Oasis will be local exploration, an optimum site could be identified, where the 20 to 50 km vicinity of the outpost site offers the greatest promise for addressing important lunar problems. If sites different from the Apollo 17 site are chosen, additional information, including high resolution (1 m) surface imaging and possibly surface surveys to determine local engineering properties, is required. The construction of the Oasis does not require the prior analysis of samples from the site chosen.

No life sciences precursor missions appear to be required in the area of space adaptation. Although there may be value in the study of the effects of weightlessness in Earth orbit in understanding the results of exposure to the 0.16-g lunar environment for extended periods of time, an iterative strategy can be adopted whereby the analysis is made as the outpost is constructed and activated. The study of crew selection and certification for extended duration assignments at the Oasis will require substantial preparatory research. In general, most of the recommendations of the NASA Life Sciences Strategic Planning Study Committee articulated in the 1988 report, *Exploring the Living Universe*, are applicable to this strategy, with less emphasis on microgravity demonstrations of medical technology and life support systems on Space Station Freedom.

Additional science objectives will be derived in detail, based on scientific progress made in the various disciplines, before the Oasis is undertaken. All prior work can be viewed as precursor information in that regard, and vigorous scientific programs should precede the Oasis, both to develop the best set of science experiments for the Oasis and to ensure that there is an adequate cadre of scientists to conduct the research that will be enabled. All science disciplines should expect to develop improved or novel experimental apparatus to take advantage of the Oasis, consistent with the transportation and operational constraints. When the Oasis is approved as a project, a portion of the development effort should be expended on the requirements of the consolidation phase of the Oasis, particularly in addressing the potential for using indigenous resources to advance scientific research and exploration.

<u>*Trade Studies*</u>. Many aspects of the Lunar Oasis require more complete understanding before development is undertaken. Some of these aspects are:

a. Comparison of the variety of ways to provide pressurized volumes for the oasis phase and beyond. Although inflatable structures have been assumed here, alternatives exist. For example, tunneling beneath the lunar surface has been considered, as has the utilization of Space Transportation System external tanks. Consideration should be given to transportation, space operations, and lunar surface operations elements of the problem.

b. Energy storage and transmission on the lunar surface. A comparison between improving the efficiency/performance of systems brought from Earth versus constructing systems based on lunar indige-

nous materials is necessary. It is assumed here that much of the mass of a photovoltaic array can be produced from lunar materials. There are still problems associated with storage of energy during the long lunar nights. One solution, potentially available in the utilization phase, is to construct a lunar photovoltaic grid at sites remote from the main facility. When expansion begins by utilizing indigenous resources, it is not required that all elements be located at one site, although additional surface transportation will be required.

c. Operations-autonomy-automation tradeoff. A study should be performed to identify approaches to operational architectures for the Oasis, which might range from autonomous activities on the Moon to activities managed from Earth, and to determine the optimal terrestrial support for lunar Oasis crews. The training/simulation/research requirements for a terrestrial analog control center/training center and its interaction with the Oasis requires study.

6.8.6 Conclusions

From the top-level assessment of this emerging case study, the Oasis appears to be a feasible program in scale. The results of this case study should be viewed as preliminary; the case could, however, be used as a baseline mission for further detailed examination as a focused case study. Assessments of the performance required from each of the production elements and their mass and energy requirements indicate that the essential elements for nearly self-sufficient operations can be delivered with the constraint of individual payloads of 20 t, and the total number of missions is not unreasonable. An organized program of technology development and scientific understanding should be initiated to better define the elements and capabilities of the Oasis. It is likely that the application of new technology and alternative approaches to the production elements can further reduce their mass and energy consumption, can provide better integrated systems, and can reduce the need for crew operations. The conduct of technology development and science missions should be an essential foundation for NASA's near-term program.

Much of the technology development envisioned here is not traditional in the space program. Such technology is better understood by elements of non-aerospace industry, including construction, chemical processing, agriculture, and others. The fact that the technology is new and that new classes of technical disciplines are included raises natural barriers to the adoption of the technologies in the space program. Studies are necessary that will allow experienced space program engineers and technical experts in a broad range of process industries to work together toward benchtop and pilot-scale plants to demonstrate to skeptics that the concepts can work efficiently and reliably. Most of this can be carried out on Earth in a 1-g environment, which means that highly relevant research and development can be initiated soon.

6.9 NEAR-EARTH ASTEROID EXPEDITION EMERGING CASE STUDY

6.9.1 Rationale

The principal reasons for considering the human exploration of near-Earth asteroids are to assess potential in situ resources and to enhance scientific understanding of the history of the solar system. The class of solar system bodies termed Aten, Apollo, and Amor asteroids have orbits outside the main asteroid belt. The Atens are asteroids with a semimajor axis less than that of the Earth. The Apollos have orbits that cross Earth's. The Amors approach Earth, coming closer than Mars, but not inside Earth's orbit. Together, these asteroids have been termed "near-Earth asteroids." These objects are generally smaller than 20 km in diameter and have a variety of compositions, as determined from reflection spectroscopy, that appears similar to that of meteorites that have fallen on the Earth. (Details supporting this statement can be found in McFadden, L. et al (1989) *Physical Properties of Aten, Apollo and Amor Asteroids*, in *Asteroids II*, University of Arizona Press.) These data suggest that the range of compounds found in meteorites, from carbonaceous water-rich materials to pure metal, may be present in asteroids. Thus, these objects have been cited in *Space Resources*, a book by J. and R. Lewis, as potential sources of materials for space construction and other economic uses. Because the asteroids are believed to be remnants of an early epoch of solar system history, they are also of great interest scientifically. The small size of these bodies and the likelihood that they have been involved in repeated impact events over their lifetimes suggest that they will be structurally complex and difficult to analyze without detailed sampling and three-dimensional analysis.

The velocity change requirements for missions from Earth to some near-Earth asteroids are small. However, the orbital characteristics of the asteroids are important. The more eccentric the orbit and the higher the inclination of the orbit to that of Earth, the higher the delta-V requirements. In addition, typical trip times for the lowest delta-V missions are on the order of 2 years and tend to be quite variable depending on the particular launch year.

From the point of view of the case study process, the near-Earth asteroids are of interest if they provide exploration targets that lead to different exploration strategies than those for planetary missions to the Moon and

Mars. For example, if the propulsion requirements for particular asteroids are small enough that round-trip missions can be accomplished in much shorter times than Mars missions, an asteroid mission could be carried out as part of the development of capability for later missions to Mars. If the trip times and propulsion requirements are similar, the asteroids become targets of opportunity that can utilize the technology developed for Mars missions; however, in the near term, such missions would not have to be studied in detail, as the concepts and designs for the Mars missions would bracket the possibilities for asteroid missions.

6.9.2 Approach

The McFadden paper referenced earlier states that the number of known objects in these groups is in the hundreds, and they are being discovered at the rate of a few each year by dedicated asteroid search programs. The Jet Propulsion Laboratory maintains a database that includes the orbital data and, where available, compositional information determined by spectroscopic studies. The database was searched for near-Earth asteroids that might have short trip times or low-energy trajectories of particular interest. Missions with trip times of 1, 2, and 3 years were then considered. Table 6.9.2-I lists the asteroids included in the study.

6.9.3 Mission Analysis

In order to analyze mission performance for round-trip asteroid missions, a standard mission concept was defined, as shown in table 6.9.3-I. This definition was used to compare initial mass requirements in low-Earth orbit for the missions defined in table 6.9.3-II. Five representative "best" missions are compared in table 6.9.3-III.

In general, the requirements for trips to the easiest targets are somewhat greater than those for round trip missions to Phobos. Table 6.9.3-IV compares the delta V requirements for a 440-day round-trip "sprint" mission to Phobos to those of the 1-year missions to 1982DB and 1989FC. Similar comparisons hold for low-energy missions, comparing opposition-class missions to the Mars system, when aerobraking is assumed for the capture at Mars. Round trips to 1982DB can be carried out for substantially less delta V than round trips to the lunar surface.

6.9.4 Conclusions

Although near-Earth asteroids pass within the orbital distance of the Earth from the Sun, simple missions to these targets do not abound. No mission was discovered that appeared to offer a near-term alternative to Mars missions in terms of reducing round-trip times for reasonable initial masses in low-Earth orbit. In general, missions to Mars orbit are similar to and better than missions to near-Earth asteroids in terms of spacecraft performance requirements. It is concluded that the work done by the Office of Exploration on missions to Mars and Phobos is sufficient to understand the systems approaches to the asteroid missions.

It is possible that new near-Earth asteroids with more accessible orbits will be discovered in the future. The capability exists to rapidly determine the mission characteristics of a round-trip manned or unmanned mission for any newly observed asteroid. The search for new asteroids should be continued.

This study has not addressed the suitability of near-Earth asteroids as targets for unmanned missions or missions using advanced propulsion systems. Such missions could be considered for some concepts of asteroid material utilization. A preliminary analysis of nuclear electric propulsion missions to near-Earth asteroids was reported in a 1983 paper by Science Applications, Inc., entitled *A Quick-look at NEP Returned-Mass Capability for Near-Earth Asteroid Missions*, in *Defense Applications of Near-Earth Resources Workshop*, University of California at San Diego, La Jolla, California. That study indicated that positive mass payback ratios could be obtained from many asteroids using advanced propulsion capabilities.

TABLE 6.9.2-I.- CHARACTERISTICS OF CANDIDATE NEAR-EARTH ASTEROIDS

	Type*	Family	Diameter (km)	Period (years)	Eccentricity	Perihelion (AU)	Inclination (deg)
1982 DB	Unknown	Apollo	1.0	1.82	0.36	0.953	1.42
1977 VA	C	Amor	1.6	2.55	0.39	1.129	2.98
Anteros	S	Amor	4.0	1.71	0.26	1.064	8.71
Orpheus	Unknown	Apollo	0.4	1.33	0.32	0.819	2.69
1989 FC	Unknown	Apollo	Unknown	1.04	0.36	0.654	4.98

* C-type asteroids are believed to be similar to carbonaceous meteorites. S-type asteroids have spectra similar to those of stony meteorites.

TABLE 6.9.3-I.- MISSION DESIGN INFORMATION FOR SEVERAL OPPORTUNITIES TO SELECTED ASTEROIDS

Asteroid	Round-trip flight time (years)	Year depart	Earth escape delta V (km/sec)	Delta V rendezvous (km/sec)	Departure delta V (km/sec)	Return delta V* (km/sec)
1982 DB	0.936	2010	3.871	5.498	5.146	0.000
	0.999	2000	4.711	6.573	1.862	0.000
	1.970	2000	4.284	0.869	0.582	0.000
	2.021	2011	4.342	1.583	1.770	0.000
	2.089	2009	4.382	3.456	2.033	0.000
	2.140	2001	4.588	0.955	0.668	0.000
	2.700	2004	4.399	0.761	2.317	0.000
	3.877	2004	4.395	0.741	1.558	0.000
	3.914	2011	4.166	0.408	1.504	0.000
	4.092	2018	4.479	1.099	0.133	0.000
Anteros	0.959	2008	4.823	4.927	5.135	0.000
	0.999	2013	4.377	4.832	4.947	0.000
	2.001	2000	4.348	5.251	4.857	0.000
	2.001	2002	5.591	2.313	4.900	0.000
	2.001	2007	4.399	3.969	4.316	0.000
	2.001	2009	4.821	2.645	5.974	0.000
	2.992	2004	4.975	1.280	1.256	0.000
1977 VA	0.958	2004	4.632	8.568	6.554	0.000
	0.999	2005	3.303	5.928	9.051	0.000
	1.988	2003	4.439	1.840	1.576	0.000
	2.009	2008	4.929	3.573	2.268	0.000
	2.010	2000	5.134	1.936	3.437	0.000
	2.382	2007	4.576	9.011	5.180	0.000
	2.526	2010	3.229	6.807	4.246	0.000
	2.566	2002	3.738	6.551	4.976	0.000
1989 FC	1.325	2005	4.952	2.813	2.728	0.000
	1.408	2009	3.934	3.726	2.884	0.896
	1.421	2008	4.078	3.910	2.216	0.896
	1.475	2015	4.926	1.992	0.235	1.910

* Impulsive velocity change required to slow spacecraft to hyperbolic excess of 8.5 km/sec.

TABLE 6.9.3-II.- MASS PERFORMANCE MODEL FOR ASTEROID MISSIONS

Number of crew	4
Crew consumables (kg/day)	7.337
Crew and crew equipment mass (kg)	1,200
Crew recovery capsule mass (kg)	6,000
Aerobrake shield factor (kg/kg entry mass)	0.15
Interplanetary mission module (kg)	30,000
Large engine mass (kg)	1,588
Small engine mass (kg)	910
Propellant boiloff factor	0.100
Earth to target navigation delta V (km/sec)	0.050
Target proximity maneuver delta V (km/sec)	0.100
Target to Earth navigation delta V (km/sec)	0.050
Propellant tankange factor (kg/kg propellant)	0.12
Small delta V system Isp (sec)	316
Large delta V system Isp (sec)	470
Number of small engines	2
Number of large engines	6

TABLE 6.9.3-III.- COMPARISON OF FIVE REPRESENTATIVE MISSIONS

Mission Target	Round-trip duration (years)	Launch year	Initial mass in LEO (t)
1982 DB	1.0	2000	5.633×10^3
Anteros	1.0	2013	4.877×10^3
1989 FC	1.4	2008	1.286×10^3
1989 FC	1.5	2015	0.649×10^3
1982 DB	2.0	2000	0.360×10^3

TABLE 6.9.3-IV.- COMPARISON OF ROUND-TRIP MISSION REQUIREMENTS

Mission Target	Round-trip flight time (years)	Year depart	Earth escape delta V (km/sec)	Delta V rendezvous (km/sec)	Departure delta V (km/sec)	Return delta V* (km/sec)	Total delta V (km/sec)
1982 DB	0.999	2000	4.711	6.573	1.862	0.000	13.146
1989 FC	1.475	2015	4.926	1.992	0.235	1.910	9.063
Phobos	1.205	2002	4.200	1.200	2.800	0.000	8.200

* Impulsive velocity change required to slow spacecraft to hyperbolic excess velocity of 8.5 km/sec.

SECTION 7

Conclusions

The goal of the FY 1989 Exploration Studies was to develop the database from which subsets of different case studies could support a defined approach. This section summarizes the key findings that highlight this year's studies.

7.1 MARS TRAJECTORY OPTIONS

This year's studies emphasized two major classes of Mars trajectories. The first, opposition, is a class of trajectories with a short stay time in the Mars system (about 30 days), characterized by short round-trip flight times (about 500 days); however, each one-way leg can be 200 to 350 days long, depending upon launch opportunity. A Venus gravity assist is typically employed for energy reduction on either the outbound or inbound leg, depending upon the launch opportunity. If a Venus gravity assist is not employed, the trajectory still passes by the Sun at Venus's orbital distance (0.7 au). A "sprint" trajectory is a subset of this class, characterized by round-trip flight times of 400 to 450 days. This class of trajectory is suitable for expedition missions, where short Mars stay-times and relatively quick round-trip missions are desired.

A second class of trajectories, conjunction, has "long" Mars stay time (about 600 days), characterized by "long" round-trip flight times (about 1,000 days); however, each one-way leg can be as short as 100 to 150 days long, depending upon opportunity. Minimum-energy trajectories are of this type; however, the one-way trip times can be as long as 300 to 350 days (total mission duration is still about 1,000 days). Reducing one-way trip times from 300 to 150 days can be achieved for a very modest (about 5 percent) increase in TMI delta V. This class of trajectory is suitable for missions where a long Mars surface stay-time, along with a large amount of mass to the Mars surface, is required. Cargo missions employ this class of trajectory for one-way missions to Mars.

From these two broad classes of trajectories, a subset with a free-return abort capability was developed for all piloted missions. This means that once the TMI burn is performed, the piloted transfer vehicle can return to Earth without any further propulsive maneuvers required.

Last year's (FY 1988) studies employed a split/sprint mission strategy for the human expeditions to Phobos and Mars. This strategy assumed that the massive TEI stage, along with all the mass destined for the Mars surface, was sent to Mars on a minimum energy cargo mission, separate from the crew, which launched during the next opportunity on a fast "sprint" profile with a total round-trip mission duration of 440 days. For mission safety reasons, this year's studies assumed that the TEI stage could not be separated from the crew's transfer vehicle. Also, by extending the sprint trajectory total mission duration to approximately 500 days, the energy requirements were comparable to the 1,000-day class requirements. The combination of these two factors obviated the need to "split" the mission, and hence eliminated the added mission operational difficulty of propellant storage and transfer in Mars orbit. Therefore, the split/sprint strategy was not used for this year's Mars Expedition case study.

Expedition versus Evolution - Trajectories for Mars missions are distinguished by the length of the stay time available at Mars. In general, these trajectories fall into the following two classes: (1) 30- to 100-day stay, and (2) 500- to 600-day stay. (Although these ranges of stay time vary by opportunity and propulsion system employed, there are two distinct options separated by a region of non-availability.) Each class of trajectory supports a different mission scenario: short stay-times for expeditionary missions, and long stay-times for the evolutionary missions.

Expeditions - These missions are characterized by short stay times at Mars, and relatively short round-trip missions. This mission strategy is appropriate for the first several missions to the martian surface. The crew's surface time and total mission duration are minimized consistent with a conservative strategy for the first piloted mission to another planet. Surface tasks conducted by the crew include reconnoitering the landing region for final site selection of the surface outpost infrastructure.

Evolution - The evolutionary strategy employed during the FY 1989 studies assumed that once the outpost site has been selected, the surface infrastructure has been delivered and deployed, and human long duration habitability issues have been resolved at the lunar outpost, a Mars crew will be committed to a long duration stay at the Mars outpost. Long duration stays on the surface are a more efficient strategy for the human resource given the capital investment in the IMLEO for a piloted Mars mission. (Expedition missions require 800-1,000 t IMLEO versus 600 to 750 t for long duration stay missions.)

An important lesson learned from the Lunar Evolution case study in this regard was the fact that long crew rotations (i.e., long surface stay time) promoted outpost expansion. With fewer flights required to transfer crews, those same resources can be applied to delivering infrastructure to the lunar surface.

7.2 PROPULSION SYSTEM TRADES

7.2.1 Mars Missions

Cryogenic/Aerobraking - The initial mass required for an all-propulsive mission using cryogenic chemical (LOX/LH_2, Isp~470 sec) propulsion is prohibitive (1,500 to 2,000 t per mission). Using aerobraking at both Mars and Earth return reduces this mass requirement by up to 50 percent. Cryogenic propulsion with aerobraking could require a lower development cost because the technology may be closer at hand. However, it will require orbital assembly of a large aerobrake, with rigorous on-orbit verification requirements.

Solid Core Nuclear Thermal Rockets - All-propulsive missions using solid core nuclear thermal rockets have initial mass in LEO requirements comparable to cryogenic with aerobraking for equivalent trip times. Nuclear thermal rockets have both high thrust and high Isp, and could be a level of investment comparable to cryogenic with aerobraking, given the NERVA development program and test firings conducted in the 1960s. This option, however, introduces the issue of nuclear power in LEO and the associated operational difficulties of humans in proximity to a nuclear source. Solid core nuclear thermal rockets in combination with aerobraking can reduce the IMLEO of both the all-propulsive solid core nuclear thermal rockets and cryogenic with aerobraking by up to 40 percent; however, it may be operationally impractical to aerobrake a hot nuclear reactor back into Earth orbit.

Gas Core Nuclear Thermal Rockets - The technology progression from solid core nuclear thermal rockets is to gas core nuclear thermal rockets, which have high thrust and Isp of 2,500 to 5,000 sec. Gas core nuclear thermal rockets provide the option of either: (a) IMLEO of approximately half the cryogenic/aerobraking requirement for the same trip time, or (b) very quick trip times (<100 days each way) with IMLEO comparable to cryogenic/aerobraking. However, it is a very immature technology requiring significant basic research and development. Also, open loop gas core nuclear thermal rockets release radioactive effluent in the exhaust plume, thereby creating operational problems in LEO. The potential advantages of nuclear thermal rocket propulsion for piloted Mars missions make this an attractive subject for further study.

Electric Propulsion - Mars missions employing electric propulsion, both nuclear (NEP) and solar (SEP), are less sensitive to launch dates and launch windows due to the low-thrust characteristics of the propulsion systems. Trip times for either system can be comparable to cryogenic/aerobraking, given a high enough power input to the thrusters. For SEP, this means solar arrays the size of several football fields; for NEP it means space nuclear reactors in the multi-megawatt class. Also, due to the low-thrust nature of the propulsion system, each would need to be operated from high-Earth orbit in order to achieve the competitive trip time. This may make electric systems impractical for piloted missions, due to the length of time the crew will spend in the Van Allen radiation belts. However, the propulsive efficiency of the high Isp of electric propulsion systems (2,000 to 10,000 sec) makes these systems attractive for cargo missions, which have higher mass requirements and are also less sensitive to flight time.

Unconventional Systems - The use of a momentum exchange facility (tether) at Phobos was included in the original mission description for the Mars Evolution case study. By placing a tether on Phobos, a vehicle can begin at Phobos and be reeled out, either toward or away from Mars. By selecting the proper length for the tether, a vehicle can descend from Phobos to the surface of Mars without performing a de-orbit delta V. Likewise, upon Mars departure, the tether can be used to add energy to an Earth-bound vehicle without consuming propellant, thereby reducing the TEI delta V.

However, Phobos is in a circular, equatorial orbit about Mars. In order to rendezvous with Phobos, the MPV must perform a delta V maneuver to raise the post-aerocapture orbit periapsis out of the martian atmosphere to an altitude equivalent to Phobos's orbital radius. A plane change maneuver must also be performed in order to place the MPV into an equatorial orbit. When adding the additional mission complexity and delta V costs to the total transportation budget, the net effect is that the gain of using the tether facility is actually offset by the cost of getting to Phobos to use it. (This is true even if the plane change maneuver is performed via aerocapture.) When the tether system masses (75 t) are accounted for, the initial mass in LEO for a transportation scheme using a Phobos tether is increased even more compared to a transportation system that does not use the tether. Therefore, this concept was excluded from the MASE integrated mission description.

7.2.2 Lunar Missions

Cryogenic/Aerobraking - The initial mass requirements for an all-propulsive mission using cryogenic chemical (LOX/LH_2, Isp~480 sec) propulsion can be reduced by 15 to 25 percent by using aerobraking at Earth return.

Nuclear Propulsion - Because of the short trip times to the Moon, and the operational considerations of nuclear systems in the Earth/Moon space, nuclear propulsion options have not been explored in depth for lunar missions.

7.3 VEHICLE DESIGN ALTERNATIVES

Expendable versus Reusable Spacecraft - Employing reusable vehicles is predominantly driven by economic considerations; i.e., reusing spacecraft permits fewer copies of the vehicle to be manufactured and launched into orbit. However, reusing spacecraft requires facilities located in space to store, maintain, and refurbish these vehicles, and the vehicles themselves must be designed to be space-based and maintained. Presumably, there is a flight rate at which reusing vehicles becomes operationally and economically a more viable option than expending vehicles after each mission. Spacecraft reusability is also dependent on the mission. Mars excursion vehicles, for example, are not practically considered reusable but instead, employ two-stage expendable designs. The martian atmosphere and sizeable gravity field (3/8 of Earth's) make single-stage reusable designs impractical.

The lunar transportation system baselined for the Lunar Evolution case study is completely reusable, with 10 mission vehicle lifetimes. However, initial lunar excursion vehicle expendability in the Lunar Evolution case study allows early emplacement of a significant manned facility. Also, larger transfer vehicles allow the delivery of fully fueled excursion vehicles to lunar orbit, which minimizes the on-orbit refurbishment and processing, reduces the number of LTV flights for each LEV landing, and postpones complicated operations until more experience has been gained.

For the Mars spacecraft, the trans-Mars injection stage is expended after each mission due to the energy required to return the spent stage to Freedom. The Mars transfer vehicle is reusable, as it returns the crew to LEO. The Mars excursion vehicle is expendable.

Zero-g versus Artificial-g Mars Vehicle - Both zero-g and artificial-g Mars vehicles have been studied in recent years. Results from these studies have provided in-depth understanding of the design of both types of vehicles. These studies have shown that artificial-g Mars transfer vehicles are on the order of 30 percent more massive than an equivalent zero-g vehicle and are more complex.

Life sciences research has not yet yielded sufficient understanding of the zero-g/artificial-g issues. Countermeasure technology and techniques will need to be developed, and, in fact, artificial-g transfer vehicles may create additional human factors problems.

In addition, fast trip times to Mars utilizing advanced propulsion methods may negate the need, if required, for artificial-g vehicles.

Propellant Transfer versus Tank Transfer - Fueling on-orbit is required if vehicles are reusable or if the entire vehicle stack (vehicle, propellant, and payload) cannot be launched fully fueled. The trade between propellant transfer and tank transfer is dependent on the vehicle design and operation requirements and capabilities.

Propellant that is transported to low-Earth orbit must be carried in tanks that are eventually expended (refurbishment of the tanks, although possible, is not being considered at this time). Considerable losses may occur during propellant transfer operations, including transfer losses, boiloff, and propellant residuals that cannot be transferred. Tank set transfers exhibit only boiloff losses, but may increase the complexity of the operations, the vehicle dry mass, and the need for integrity and verification checks.

The current scenarios under investigation utilize both propellant and tank transfers in the fueling operations of the transportation vehicles. Fueling of the lunar transfer vehicles is accomplished via tank transfers in low-Earth orbit, whereas the lunar excursion vehicles are fueled through propellant transfers in low lunar orbit. Both tank and propellant transfers are used for the Mars vehicles.

7.4 PLANETARY SURFACE SYSTEM OPTIONS

Open versus Closed Loop Life Support - The life support system selected for the evolutionary outposts is an advanced regenerable physical-chemical system that evolves in two phases. The initial life support system is a derivation of the Space Station Freedom system and recovers 96 percent of the water and oxygen throughout the system. However, nitrogen, food, and system and crew expendables are still required for resupply. The second phase of the system advances the recovery of materials from solid wastes and reduces resupply of system expendables. This system also uses lunar-supplied oxygen in the regenerable system, achieving a functional closure of nearly 99 percent. Resupplies are still required for nitrogen, food, and crew expendables. However, the net savings in mass to the lunar surface (for example) is more than 60 percent over a 20-year period, versus an open-loop (Shuttle-class) system, as shown in figure 7.4-1.

In Situ versus Earth-Supplied Resources - The use of in situ resources reduces the logistical demands on Earth of maintaining a lunar outpost and helps to develop outpost operational autonomy from Earth. The MASE Controlled Trade Study conducted this year on this topic determined that lunar resource production results in less total mass to LEO over the life of a lunar outpost, and also produces a strong potential for return on investment, as shown in figure 7.4-2. Prior studies have demonstrated that an extensive terrestrial experience base

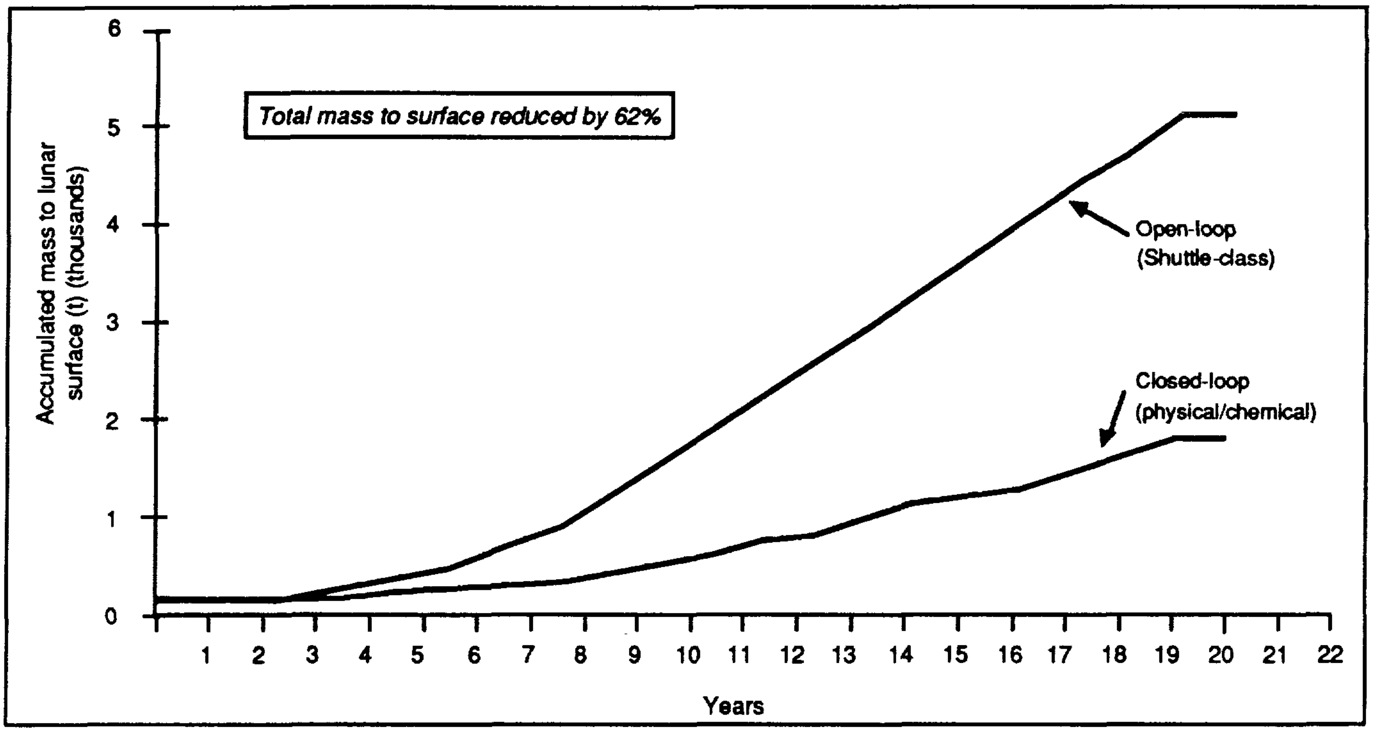


Figure 7.4-1.- Accumulated mass to the lunar surface for closed and open life support systems.

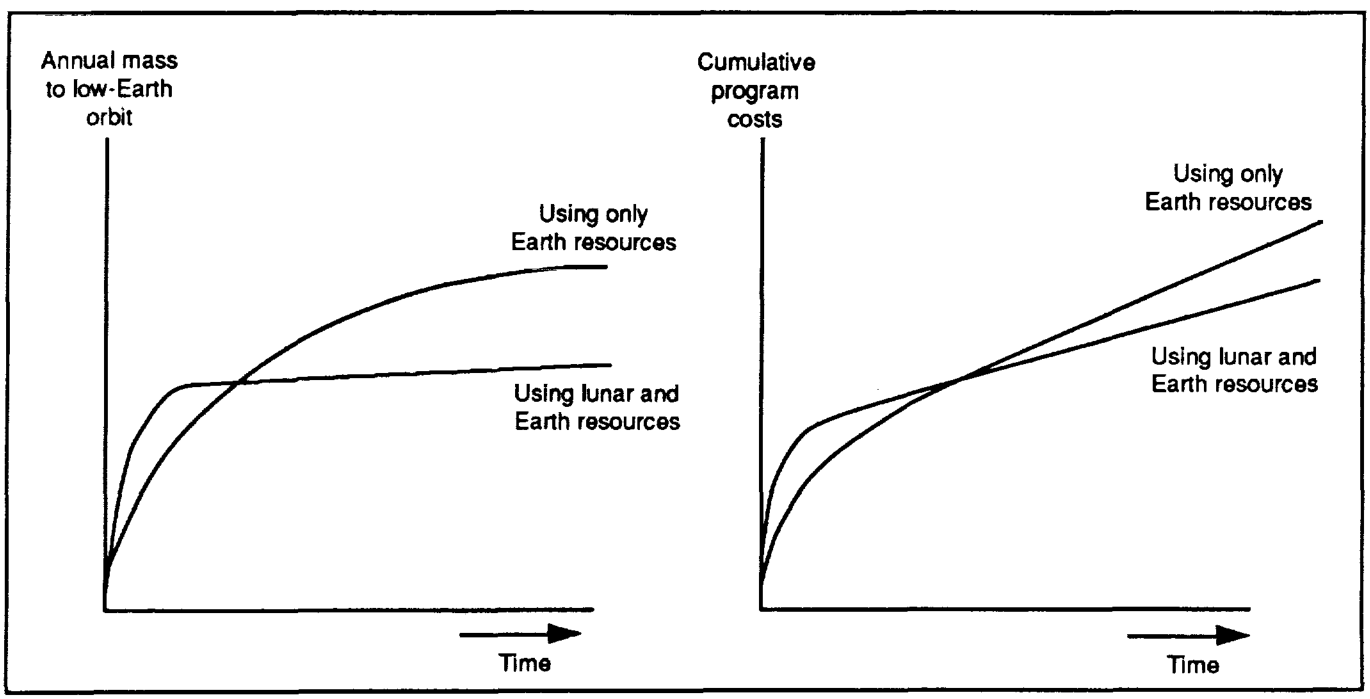


Figure 7.4-2.- Effect of in situ resource utilization on mass and cost for lunar missions.

with analogous mining processes exist. In fact, the terrestrial processes use highly automated/teleoperated facilities with production rates orders of magnitude greater than lunar requirements. Results to date have shown that for the Lunar Evolution case study the initial mass in LEO is reduced by as much as 35 percent per lunar flight when LLOX is used in the lunar ascent/descent transportation systems.

The Mars Evolution case study assumed the development of a propellant production plant at Phobos. The case study transportation architecture assumed the propellant produced from the Phobos regolith was used by the MPV for the TEI delta V. The TEI propellant mass required, for the spacecraft masses and operational sequences assumed, varies from 60 to 80 t per return flight depending on the opportunity. This translates into a mass savings in LEO of 200 to 300 t per piloted flight.

However, these mass savings are not completely realized. First, there is the mass cost of delivering the 50 t plant to Phobos initially, plus any spares that must be delivered over the life of the plant. Next, there is the additional delta V (and hence, mass) cost of actually getting to Phobos, compared to operations out of a highly elliptical orbit, as discussed in section 7.2.1.

The net result is that there is a positive mass return from a propellant plant at Phobos. However, the magnitude of the return realized was not large enough to offset the increased mission operational complexity in getting to Phobos to use the plant. Also, it was assumed in the mass calculations that once the Phobos plant was on-line, the MPV departed Earth with zero TEI propellant. From a crew and mission safety standpoint this was not felt to be practical. Accounting for these concerns would decrease (or perhaps obviate) the mass benefit further. Furthermore, since Phobos is in an equatorial orbit, rendezvous with Phobos would constrict the MEV to only equatorial sites on Mars without the additional penalties associated with plane change maneuvers.

It was therefore concluded in the synthesis activity that the use of Phobos as a propellant gateway was not a viable option. However, this conclusion is dependent upon the case study assumptions about how and where the Phobos plant was utilized. For example, the scope of the Mars Evolution case study extended only through eight flights. Extending the horizon another 10 to 20 years may produce very different results. Delivering the Phobos-produced propellant to the MPV in its equatorial orbit instead of delivering the MPV to Phobos would change the results dramatically. Also, using Phobos-produced propellant in the MEVs will likely produce a very different result, since it is expected that, like the use of LLOX in the Lunar Evolution case study, the benefit realized is directly proportional to the proximity to the source.

Surface Power - During the emplacement phase of the Lunar Evolution case study, a solar power system was selected as the major power source. The system uses photovoltaic arrays (PVA) for daytime power and regenerative fuel cells (RFC) for power during the lunar night. Due to the length of the lunar night (14.8 days), the storage requirements for the RFC are massive. The initial PVA/RFC system provides 50 kW during the day and 25 kW during the night. As the power demands grow, the solar array systems become unwieldy. Above 100 to 200 kW, nuclear power offers improved specific power. In fact, a major finding documented in the Lunar Evolution case study is that using nuclear power to provide 100 kW continuously for 10 years will reduce the mass to the lunar surface by more than 90 percent compared to a PVA/RFC system.

7.5 ORBITAL NODE OPTIONS

Space Station Freedom versus New Spaceport versus Direct Assembly - The use of on-orbit assembly for this year's case studies ran the gamut from: (a) no assembly (Mars Expedition) to (b) assembly at Space Station Freedom (Lunar Evolution) to (c) assembly at a free-flying fixture (Mars Evolution). Each was proposed to understand the implications of the respective concept, not to recommend one option over the others as a final implementation. The key results of the respective studies are presented in table 7.5-I.

7.6 SUMMARY

Over the last 2 years, NASA has examined in detail a number of potential strategies for human exploration of the solar system: Apollo-type expeditions to Mars and Phobos, evolutionary outposts on the Moon and Mars, and scientific observatories on the Moon. The case studies, controlled trades, and special assessments have resulted in a substantial base of information about the benefits and implications of various approaches to expanding human presence and activity beyond Earth's orbit into the solar system. The key findings of the 1989 exploration studies combine with those of the 1988 studies to build the foundation of future analyses to determine how best to achieve this national goal.

TABLE 7.5-I.- LOW-EARTH ORBIT NODE OPTIONS

	Advantages	Disadvantages
No LEO node (vehicle "self assembly")	• Minimizes on-orbit facilities • No impact to Space Station Freedom • Minimum crew required for assembly	• Burdens transfer vehicle designs with the overhead of "self assembly" • No vehicle reusability capability • Drives up the size of ETO vehicles
Space Station Freedom co-orbiting platform	• Assembly facility in close proximity to Space Station Freedom • Relieves "self assembly" from vehicles • Minimal impact to Space Station Freedom • Space Station Freedom crew available for assembly	• Requires crew transportation between Space Station Freedom and co-orbiting facility • Formation flying complexity
Space Station Freedom (multi-purpose)	• Single common facility in LEO for research and vehicle assembly • Minimizes logistics required • Crew available for assembly	• Interferes with Space Station Freedom research
Separate LEO assembly facility	• Optimized on-orbit facilities • No impact to Space Station Freedom • Separate crew available for assembly	• Two similar facilities in LEO • Dual logistical interfaces

APPENDIX

LEXICON

LEXICON

This report uses a number of acronyms, abbreviations, and special terms. In order to facilitate the reader's comprehension of the text, a lexicon is presented here.

ACRONYMS AND ABBREVIATIONS

A&R/HP - automation and robotics/human performance
Ab - aerobrake
AB - aerobraking
AC - assembly complete (referring to Space Station Freedom)
ADAB - allowable dose above background
AFE - Aeroassist Flight Experiment
AI - artificial intelligence
ALARA - as low as reasonably achievable (radiation dose limit guidelines)
ALS - Advanced Launch System
alt - altitude
Ant - antenna
AOTF - acousto-optical tunable filter
AOTPM - ascent and orbit transfer propulsion module
AP - all-propulsive
APM - ascent propulsion module
APSA - advanced photovoltaic solar array
ARC - Ames Research Center
Artificial-g - artificial gravity
ASAO - Advanced Space Analysis Office (located at the NASA/Lewis Research Center)
a-Si - antimony and silicon
ASRB - Advanced Solid Rocket Booster
ASRM - Advanced Solid Rocket Motor
ATCS - Active Thermal Control Subsystem
ATD - Advanced Technology Development
ATDRSS - Advanced Tracking and Data Relay Satellite System
ATP - authority to proceed
au - astronomical unit

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bps - bits per second
BTF - booster turnaround facility

C - Centigrade or Celsius
C_3 - square of the hyperbolic excess velocity in units $(km/s)^2$
C&C - command and control
C&T - communications and tracking
CAB - core assembly building
CAD/CAM - computer-aided design/computer-aided manufacturing
CBC - closed Brayton cycle
CC - command and control
CCD - charge coupled device
CD - coefficient of drag
CDR - Critical Design Review
CELSS - controlled ecological life support system
CEPF - cargo element processing facility

CEPS - Center Exploration Program Scientists
CERTS - Crew Emergency Return Transfer Stage
CERV - Crew Emergency Return Vehicle
CFES - continuous flow electrophoresis system
CG - center of gravity
CIF - cargo integration facility
CM - command module
cm - centimeter
CMG - control moment gyro
CO_2 - carbon dioxide
Code C - Office of Commercial Programs
Code E - Office of Space Science and Applications
Code EB - Office of Space Science and Applications, Life Sciences Division
Code EL - Office of Space Science and Applications, Solar System Exploration Division
Code M - Office of Space Flight
Code N - Office of Management
Code R - Office of Aeronautics and Space Technology
Code S - Office of Space Station
Code T - Office of Space Operations
Code Z - Office of Exploration
CSF - customer servicing facility (on Space Station Freedom)
CSI - control/structure interaction
CSTI - Civil Space Technology Initiative
cu - cubic
CY - calendar year

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d - days
D - diameter
dB - decibel
dc - direct current
DDT&E - design, development, test, and evaluation
DIPS - dynamic isotope power system
DMS - data management system
DoD - Department of Defense
DoE - Department of Energy
DSC/EGA - differential scanning calorimeter/evolved gas analyzer
DSN - Deep Space Network
ΔV - delta velocity or mission velocity increment in units of kilometers/second (km/s)

ECCV - Earth crew capture vehicle
ECLSS - environmental control and life support system
EDCO - extended duration crew operations
EDO - extended duration orbiter
EM - electromagnetic
EMG - Exploration Management Group
EMU - extravehicular mobility unit
EOC - Earth orbital capture
EOD - Exploration Opportunities Document
EP - electric propulsion
EPS - electrical power system
ERD - Exploration Requirements Document

ESA - European Space Agency
ET - external tank
ETF - Exploration Task Force
ETHPF - external tank hazardous processing facility
ETO - Earth to orbit
EVA - extravehicular activity
EXSWG - Exploration Science Working Group
EXTWG - Exploration Technology Working Group

F - Fahrenheit
FDI&R - fault detection, isolation and recovery
FDM - frequency division multiplexing
FEL - first element launch (Space Station Freedom)
FMS - fluid management system
FPSE - Free Piston Stirling Engine
FSAS - fluid services accommodations subsystem
ft - feet
F/W_i - thrust-to-initial-weight ratio
FY - fiscal year

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g - gram
g - Earth's gravitational acceleration at sea level
GCR - galactic cosmic radiation
GEO - geostationary Earth orbit (also geosynchronous Earth orbit)
GHz - gigahertz (one billion cycles per second)
GN&C - guidance, navigation, and control
GN_2 - gaseous nitrogen
GO_2 - gaseous oxygen
GPHS - general purpose heat source
GPS - global positioning satellite
GRS - geosynchronous relay satellite
GSE - ground support equipment
GSFC - Goddard Space Flight Center
GT - ground terminal
G_z - artificial gravitational acceleration

H_2 - hydrogen
H_2O - water
HEO - high-Earth orbit
HER - high expansion ratio
HLLV - heavy-lift launch vehicle
HMF - Health Maintenance Facility
hp - periapsis altitude
HPF - hazardous processing facility
HQ - Headquarters
hr - hour
ht - heat
ht - height
H/W - hardware

IA - Integration Agent
IDEAS - NASA Integrated Design and Evaluation of Advanced Systems computer software
IHPF - integrated hazardous processing facility
IHPPF - integrated hazardous payload processing facility
ILC - initial launch capability
IMLEO - initial mass to low-Earth orbit
IMM - interplanetary mission modules
IMU - inertial measurement unit
InP - indium phosphide
IOC - initial operational capability
IR - infrared
I_{sp} - specific impulse (in units of seconds)
IT&V - integrated test and verification
ITL - integrate/transfer/launch
IVA - intravehicular activity

JEM - Japanese Experiment Module
JPL - Jet Propulsion Laboratory
JSC - Lyndon B. Johnson Space Center

K - temperature in kelvins
Kbps - kilobits per second
kg - kilogram
klbf - thousand-pound force
klbs - thousand pounds
km - kilometers
kN - kilonewton (1,000 newtons)
KSC - John F. Kennedy Space Center
kW - kilowatts
kWe - kilowatts of electrical power
kWt - kilowatts of thermal power

L1/L2/L5 - Earth-Moon colinear libration points
LANL - Los Alamos National Laboratory
LAPM - lander/aerobrake propulsion module
LaRC - Langley Research Center
lb - pounds
lbm - pounds-mass
LCC - launch control center
L/D - lift-to-drag ratio
LDEF - Long Duration Exposure Facility
LEO - low-Earth orbit
LEP - laser electric propulsion
LeRC - Lewis Research Center
LEV - lunar excursion vehicle
LEV-C - lunar excursion vehicle - cargo
LEV-P - lunar excursion vehicle - personnel
LGRF - low gravity research facility
LH_2 - liquid hydrogen

Lidar - light detection and ranging
LLO - low-lunar orbit
LLOX - lunar liquid oxygen
low-g - low gravity
LOX - liquid oxygen
LP - logistics platform
LPT - lunar propellant tanker
LRB - liquid rocket booster
LRS - lunar relay satellite
LS - lunar surface
LSS - life support system
LST - lunar surface terminal
LTV - lunar transfer vehicle
LTV-C - lunar transfer vehicle - cargo
LTV-P - lunar transfer vehicle - personnel

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m - meters
M - million
Magsail - magnetic sail
MASE - Mission Analysis and Systems Engineering (JSC Code Z support function)
MAV - Mars ascent vehicle
Mbps - megabits per second
MCC - midcourse correction
MCL - Mars cargo lander
MCV - Mars cargo vehicle
MDV - Mars descent vehicle
μEP - microwave electric propulsion
MEV - Mars excursion vehicle
MeV - megaelectronvolts
MEV-C - Mars excursion vehicle - cargo
MEV-P - Mars excursion vehicle - personnel
MHz - megahertz (one million cycles per second)
MJ - multijunction
MLI - multilayer insulation
MLP - mobile launch platform
mm - millimeter
MMH - monomethylhydrazine
MMIC - monolithic microwave integrated circuits
MMV - Mars maneuvering vehicle
MOC - Mars orbital capture
MOCS - Mars orbit capture system
MOD RTG - modified radioisotope thermoelectric generator
MOI - Mars orbit insertion
MOMV - Mars orbital maneuvering vehicle
MOV - Mars orbiter vehicle
Mpl - mass of payload
MPV - Mars piloted vehicle
MRSR - Mars Rover/Sample Return
M sail - mass of solar sail
MSC - Mobile Servicing Center (an element of Space Station Freedom)
MSFC - George C. Marshall Space Flight Center
MTBF - mean time between failures
MTC - man-tended capability

μTP - microwave thermal propulsion
MTTR - mean time to repair
MTV - Mars transfer vehicle
MTV-C - Mars transfer vehicle - cargo
MW - megawatts
MWe - megawatts of electrical power
MWt - megawatts of thermal power

N - Newtons
N_2 - nitrogen
N_2O_4 - nitrogen tetroxide
NA, N/A - not applicable
NASA - National Aeronautics and Space Administration
Nascom - NASA communications network
NASTRAN - NASA Structural Analysis
NCRP - National Council on Radiation Protection
NEP - nuclear electric propulsion
NERVA - Nuclear Engine for Rocket Vehicle Application
NIB - noninterference basis
nmi - nautical mile
NOAA - National Oceanic and Atmospheric Administration
NOCC - network operations control center
NSO - Nuclear safe orbit
NSTS - National Space Transportation System
NTR - nuclear thermal rocket

O_2 - oxygen
OAA - Opportunities Assessment Agent
OAST - Office of Aeronautics and Space Technology (Code R)
OEXP - Office of Exploration (Code Z)
O/F - oxidizer-to-fuel ratio
OMB - Office of Management and Budget
OMD - Operations and Maintenance Documentation
OMS - orbital maneuvering system
OMV - orbital maneuvering vehicle
ORNL - Oak Ridge National Laboratory
ORU - orbit replaceable unit
OSHA - Occupational Safety and Health Administration
OSO - Office of Space Operations (Code T)
OSS - Office of Space Station (Code S)
OSSA - Office of Space Science and Applications (Code E)
OTV - orbital transfer vehicle

P - power
PA, P/A - payload assist
P/C - physical-chemical
PCU - power control unit
PDR - Preliminary Design Review
PDRD - Preliminary Design Review Document
PHF - payload holding fixture

PLSS - portable life support system
PMAD - power management and distribution system
PMC - permanent manned capability
POC - proof of concept
ppm - parts per million
PRR - Preliminary Requirements Review
psia - pounds per square inch absolute
PTA - propulsion test article
PV - photovoltaic
PVA - photovoltaic array
PVA/RFC - photovoltaic array/regenerative fuel cell
PV/RFC - photovoltaic/regenerative fuel cell
PYE - personnel year equivalents

R - final mass to initial mass ratio
R&A - research and analysis
R&D - research and development
R&T - research and technology
RCS - reaction control system
rem - roentgen-equivalent man
RF - radio frequency
RFC - regenerative fuel cell
RMS - remote manipulator system
RPSF - rotational payload support facility
RSAT - relay satellite
RSRM - reusable solid rocket motor
RTG - radioisotope thermoelectric generator
RTLT - round-trip light time

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s, sec - seconds
SAA - special assessment agent
S/C - spacecraft
SDI - Strategic Defense Initiative
SDV - Shuttle-derived vehicle
SEP - solar electric propulsion
SFM - servicing facility manipulator
SGL - space-to-ground link
SI - International System of Units
SIA - station interface assembly
SMr - specific mass of rover
SNAP - systems for nuclear auxiliary power
SPDMS - special-purpose dextrous manipulator system
sq - square
SRD - Study Requirements Document
SRR - system requirements review
SSES - Solar System Exploration Subcommittee
SSME - Space Shuttle Main Engine
SSPA - solid state power amplifier
SSPF - Space Station processing facility
STBE - Space Transportation Booster Engine
STEP - Structural Technology Experiments Platform
STME - Space Transportation Main Engine

STP - solar thermal propulsion
STS - Space Transportation System
STV - space transfer vehicle
Sv - Sieverts

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t - metric ton (tonne, 1,000 kg)
TCS - thermal control system
T&DA - tracking and data acquisition
TDRSS - Tracking and Data Relay Satellite System
TE - thermoelectric
TEI - trans-Earth injection
TEIS - trans-Earth injection stage
THURIS - The Human Role in Space (computer program)
TMI - trans-Mars injection
TMIS - trans-Mars injection stage
TNIM - telecommunications, navigation, and information management
TPS - thermal protection system
TPT - transfer propellant tank
TT&C - tracking, telemetry, and command
T/W - thrust-to-weight ratio
TWTA - traveling wave tube amplifier

UHF - ultra-high frequency
ULP - universal launch pad
UN - uranium nitride
UPA - universal payload adapter
US - United States

V, vel - velocity
v - volts
VAB - vehicle assembly building
VGRF - variable gravity research facility
VIB - Vehicle IntegrationBuilding
VIMS - visual/infrared mapping spectrometer
VLF - very low frequency
VLFA - very low frequency array

W - watts
We - watts of electrical power
Whr/kg - watt hours per kilogram

XMIT - transmit
XPOND - transponder

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yr - year

Zero-g - zero-gravity

DEFINITION OF TERMS

A/B - Pre-development phase of program.

Acousto-Optical Tunable Filter - An advanced sample analysis sensor.

Aerobrake - Aerodynamic brake for use in low-density atmospheres.

Aerocapture - A technique of capturing a heliocentric spacecraft into a planetary orbit, using an aerobrake.

Aeroshell - High-drag aerodynamic-braking heat shield for returning spacecraft or planetary landers.

Aphelion - Point in a solar orbit (planet or spacecraft) closest to the center of the Sun.

Ascent and orbit transfer propulsion module - A module using chemical propellants (H_2/O_2) that is used on the Phobos/Deimos excursion vehicle, Mars ascent vehicle, and Mars descent vehicle.

Astronomical unit - The distance from the Earth to the Sun; approximately 150 million k,m.

Beneficiation - Improving the chemical properties of an ore so that metal can be recovered.

Brayton engine - Engine utilizing the Brayton cycle, a thermodynamic cycle consisting of two constant-pressure processes interspersed with two constant-entropy processes.

Bremsstrahlung - Radiation that is emitted by an electron accelerated in its collision with the nucleus of an atom.

C/D - Development phase of program.

C_3 - Injection energy; square of the hyperbolic excess velocity in units of $(km/s)^2$.

Cislunar - Of or in the region of space between Earth and the Moon.

Conjunction-class trajectory - Round-trip trajectory between two planets (e.g., Earth and Mars) requiring minimum fuel expenditure. Conjunction-class Mars missions generally have flight times slightly greater than 1,000 days.

Consolidation phase - Second phase of the Lunar Evolution and Mars Evolution case studies. The objectives of this phase include:

1. Begin to exploit the local Mars/Moon resources with the introduction of a gateway propellant plant.
2. Increase crew size, Mars/Moon surface stay time, and both human and robotic exploration capabilities.

Controlled Ecological Life Support System - A spacecraft life support system that continually recycles solid, liquid, and gaseous materials essential for human life.

Cryogenic propellant - Propellant that must be stored at very low temperatures, e.g., liquid hydrogen and liquid oxygen.

Deep space maneuver - Propulsive maneuver performed along an interplanetary trajectory.

Deep Space Network - NASA Earth-based interplanetary communications system.

Direct entry - An uninterrupted atmospheric entry/landing process that allows savings in vehicle mass requirements.

Dysbarism - Sickness caused by decompression.

Earth crew capture vehicle - Small vehicle for crew Earth orbit capture and/or Earth Entry and Landing System.

Earth/Moon libration point - (also Lagrangian point) Critical point in Earth-Moon space, where a body at rest would remain unless disturbed by an external force.

Earth orbit capture system - Earth aerobrake plus retropropulsion plus guidance and control, if required.

Earth Transfer Vehicle - Configuration of Mars spaceship for Mars to Earth transport of crew.

Earth-to-orbit vehicles - Launch vehicles such as expendable launch vehicles, Space Transportation System, and heavy lift launch vehicles.

Emerging case study - Candidate concept for a focused case study that is not yet sufficiently mature for release from the MASE analysis process.

Emplacement phase - First phase of the Lunar Evolution and Mars Evolution case studies. The objectives of this phase include:

1. Initial human landing, deployment, and check-out of the surface habitat module.
2. Local human exploration activities and regional semiautonomous exploration capability.

Enabling technology - Key high-leverage technology assumed or required for completion of a spacecraft mission as required in the case study.

Exploration Requirements Document - Publication produced by the Office of Exploration that levies the overall exploration themes and objectives to initiate the FY 1989 studies activities.

Extravehicular activity - Any human activity outside protective shirt-sleeve environment and requiring a spacesuit.

Fifth force - A highly speculative weak anti-gravity force. Recent experiments at Lawrence Livermore Lab indicate there is no fifth force in gravity.

Flight telerobotic servicer - Teleoperated robot for Space Station Freedom.

Free-abort - Type of manned round-trip Mars mission designed with an abort capability, prior to Mars arrival, requiring no propulsive maneuver at Mars flyby.

Galactic cosmic radiation - Cosmic rays from outside the solar system.

Geophysical monitoring station - Mars surface scientific station designed to:

1. measure atmospheric volatiles, composition, and structure
2. study spatial and diurnal variations and long-term temporal variations
3. monitor weather, clouds, dust storms, and dust density and composition

4. measure solar wind flux, electron impact, other cosmic particles, heat flow, and passive seismic activity.

Geostationary Earth orbit - (also geosynchronous Earth orbit) Orbit in which satellite remains over same point on surface of Earth – about 35,800 km above the equator – revolving at same angular speed as Earth.

Gravity gradient stability - Means of regulating the attitude or orientation of a spacecraft by responding to changes in gravity acting on the spacecraft.

Halo orbit - Stable spacecraft orbit about a libration point such as L1 or L2.

Heavy lift launch vehicle - Earth-to-orbit vehicle with payload lift capability greater than 90 t to low-Earth orbit.

Hooks and scars - Term used in association with Space Station Freedom. Refers to design techniques that prepare Freedom to facilitate future planned evolution; scars refers to hardware, and hooks refers to software.

Hypergolic propellant - A combination of fuel and oxidizer that ignite spontaneously on contact.

Implementation (Requirements) - Those activities that react to a set of requirements with specific analyses, studies, and trades directed toward the provision of specific products, which may include concepts, element/systems architecture, recommended configurations, and operating strategies and techniques.

Insolation - Rate of delivery of solar radiation per unit area of a planet's surface.

Integration Agents - The Office of Exploration Integration Agents are Level III implementation agents. The Integration Agents are responsible for the definition of integrated sets of elements within specific domains. The domains are:

1. Space Transportation (excluding Earth-to-Orbit)
2. Planetary Surface Systems
3. Orbital Nodes

The Integration Agents act as conceptual definition agents for elements of infrastructure that support the scenario development activities. The definition activity of an Integration Agent matures with time, beginning with basic functional descriptions, and evolving to concepts and point designs, but, in general, will not go to the level of detail expected in phase B development programs. At that point in time, the development responsibility is passed from Code Z to the responsible NASA Headquarters program office.

Interplanetary mission modules - Habitat/laboratory/logistics modules for crew in space.

L1 - Libration point; critical point in Earth-Moon space where a body at rest would remain unless disturbed by an external force.

Lander/aerobrake propulsion module - Module using chemical propellants to deliver up to 50 t payload from high-Mars orbit to Mars surface. The module is used on the Mars cargo lander, Mars descent vehicle, and the Mars crew sortie vehicle.

Low-Earth orbit - A circular orbit about Earth with an altitude of approximately 300 to 500 km.

Low-lunar orbit - A circular orbit about the Moon with an altitude of approximately 100 km.

Low-Mars orbit - A circular orbit about Mars with an altitude of approximately 250 to 500 km.

Lunar day/night - Approximately 14 Earth days each. The Moon completes one revolution about Earth in approximately 28 days.

Lunar excursion vehicle - Vehicle designed to transport crew, cargo, and propellants between low-lunar orbit (300 km circular, equatorial) and the lunar surface by the year 2000.

Lunar Observer - Robotic scientific mission to study geochemistry and climatology of the Moon.

Lunar propellant tanker - Vehicle designed to transport propellant between Earth's surface and a lunar transfer vehicle in low-Earth orbit.

Lunar relay satellite - Communications satellite in a halo orbit about L1 or L2 in the Earth-Moon system. These satellites will relay information between lunar surface terminals and other lunar relay satellites.

Lunar transfer vehicle - Vehicle for transportation between low-Earth orbit and the Moon.

Magnetic sail - A superconducting current loop that creates a magnetic dipole field to deflect the charged particles of the solar wind and obtain thrust. To be exposed to the solar wind, the magnetic sail must be operated in heliocentric space, outside the influence of the Earth's magnetic field.

Magnetotail - The portion of the magnetosphere extending from Earth in the direction away from the Sun for a variable distance of nearly 1,000 Earth radii.

Mars ascent vehicle - The vehicle that is launched from Mars surface to Mars orbit.

Mars cargo vehicle - Logistics vehicle sent to Mars for cargo staging.

Mars descent vehicle - The vehicle that de-orbits to land on Mars.

Mars excursion vehicle - Spacecraft that carries crew to Mars surface from Mars orbit.

Mars landed mission module(s) - Habitat/laboratory/logistics modules for the surface of Mars.

Mars Observer - Robotic scientific orbiter mission to Mars, planned for 1992 launch.

Mars orbit operating system - Propulsion for Mars orbit maneuvers.

Mars orbital capture system - Mars aerobrake plus retropropulsion, if required, plus guidance and control.

Mars orbiting vehicle - Vehicle configuration in Mars orbit.

Mars spaceship - The spaceship that is assembled in low-Earth orbit.

Mars transfer vehicle - Spacecraft configuration during flight to Mars.

Mission Analysis and Systems Engineering - The Mission Analysis and Systems Engineering (MASE) is a Level II implementation function of the Office of Exploration. MASE will decompose the scenario requirements into collections of top-level, functional requirements that must be accomplished by the Integration Agents (IAs). The Integration Agents will develop concepts that implement these

requirements and furnish this information to the MASE for integrated systems synthesis and total scenario option evaluation.

The MASE will also develop scenario dependent study issues for the Special Assessment Agents (SAAs) and, as results are available from the Special Assessment Agents, will assess total scenario impacts.

Monomethylhydrazine - Bipropellant fuel.

National Space Policy and Exploration Guidelines -

- The policy specifies that in conjunction with other agencies: NASA will continue the lead role within the Federal Government for advancing space science, exploration, and appropriate applications through the conduct of activities for research, technology, development, and related operations.

- Space Science - NASA, with the collaboration of other appropriate agencies, will conduct a balanced program to support scientific research, exploration, and experimentation to expand understanding of: (1) astrophysical phenomena and the origin and evolution of the universe; (2) the Earth, its environment, and its dynamic relationship with the Sun; (3) the origin and evolution of the solar system; (4) fundamental physical, chemical, and biological processes; (5) the effects of the space environment on human beings; and (6) the factors governing the origin and spread of life in the universe.

- Space Exploration - In order to investigate phenomena and objects both within and beyond the solar system, the policy states that NASA will conduct a balanced program of manned and unmanned exploration.

  - Human Exploration - To implement the long-range goal of expanding human presence and activity beyond Earth orbit into the solar system, the policy directs NASA to begin the systematic development of technologies necessary to enable and support a range of future manned missions. This technology program (Pathfinder) will be oriented toward a Presidential decision on a focused program of manned exploration of the solar system.

  - Unmanned Exploration - The policy further directs NASA to continue to pursue a program of unmanned exploration where such exploration can most efficiently and effectively satisfy national space objectives by among other things: achieving scientific objectives where human presence is undesirable or unnecessary; exploring realms where the risks or costs of life support are unacceptable; and providing data vital to support future manned missions.

Nitrogen tetroxide - N_2O_4, bipropellant oxidizer.

Normoxic - Normal concentration (21%) of oxygen in Earth's atmosphere at sea level.

Nuclear electric propulsion - Low-thrust electric propulsion, with electric power provided by nuclear reactor.

Nuclear Engine for Rocket Vehicle Application - (NERVA), nuclear thermal rocket program.

Nuclear safe orbit - Circular geocentric orbit with 700 km altitude designed to delay atmospheric entry and spacecraft nuclear reactor disintegration for several hundred years in order to reduce the danger of high-level radiation in the atmosphere.

Nuclear thermal rocket - A space propulsion concept technique in which the heat from a nuclear fission reactor is used to raise the temperature of the propellant, which is then expanded through a nozzle to provide thrust. Two types of nuclear thermal rockets have been studied: gas core and solid core.

Office of Exploration Case Studies - Studies are specific mission scenarios that execute the exploration goals according to the objective content of the themes and strategies. Each case study may contain several optional implementation approaches. The case studies will be initiative-specific; each case study and its optional implementation approaches will address a single strategy. The three case studies analyzed in FY 1989 are:

1. Lunar Evolution
2. Mars Evolution
3. Mars Expedition

Opposition-class trajectory - Round-trip trajectory between two planets (e.g., Earth and Mars) requiring a higher level of fuel expenditure than conjunction-class missions. Opposition-class Mars missions generally have flight times around 500 days.

Orbital Nodes Integration Agent - The Office of Exploration Orbital Nodes Integration Agent is responsible for the definition and integration of all systems, elements and operational procedures for the low-Earth orbit transportation node, the lunar node, the Mars node and any other space or orbiting infrastructure required to support the exploration objectives.

Penetrator - Instrumented probe for subsurface and limited surface/atmospheric science investigations.

Perihelion - Point in a solar orbit (planet or spacecraft) closest to the center of the Sun.

Photonic - Adjective form of photon, a massless particle, the quantum of the electromagnetic field, carrying energy, momentum, and angular momentum.

Photovoltaic power array - Power system operated by voltage generated as a result of exposure to visible or other radiation.

Planetary Surface Systems Integration Agent - The Office of Exploration Planetary Surface Systems Integration Agent is responsible for the definition and integration of all systems and elements and operational procedures for all infrastructure that resides on the surface of the Moon, Mars, or other planetary body.

Powered abort - Type of manned round-trip Mars mission designed with an abort capability, prior to Mars arrival, requiring a propulsive maneuver at Mars flyby or a deep space maneuver with magnitude less than the trans-Earth injection maneuver.

Precursor Requirements - Science, technology, or operational data needed as critical path information to enable selection of specific habitation site location, location/objectives of specific user surface activities, systems design options, or specific operational approaches to human exploration. Precursor data are usually obtained via robotic, highly automated missions.

Prerequisite Requirements - A technical space system performance capability necessary for the execution of one or more exploration initiatives or scenarios. Prerequisite requirements are part of the exploration study and define case study-specific technology, space system, and operational support needs at a level of detail sufficient to enable the receiving program organization to proceed with its implementation strategy: either the development of new hardware elements, the modification of previously defined or existing hardware elements, or the use of existing hardware elements in support of the multi-program initiative implementation effort.

Propellant Tank Farm - Collection of propellant tanks for on-orbit fueling of interplanetary spacecraft.

Radioisotope thermoelectric generator - Self-contained power system in which a radioisotope is used to heat one junction in a circuit containing dissimilar metals, thus generating sustained electricity.

Regolith - The layer rock or blanket of unconsolidated rocky debris of any thickness that forms the surface of much of the Moon.

rem - A unit for measuring absorbed doses of radiation.

Remote Manipulator System - Space shuttle robot arm.

RL-10B - LH_2/LOX engine, manufactured by Pratt & Whitney.

Shuttle-C - Space Shuttle derivative proposed unmanned cargo vehicle.

Shuttle-Z - Space Shuttle derivative operating from Shuttle launch pads and capable of transporting a fully assembled Mars spacecraft into low-Earth orbit in one or two launches.

Sol - A mean solar day for a given planet. Martian day; the modern term for the rotation period of Mars: 24h 37m 22.6s.

Solar electric propulsion - Ion drive; solar power; utilized in rocket systems; based on electric power, which can be derived from solar cells.

Solar particle event - Eruption from the Sun's chromosphere, which may appear within minutes and fade within an hour. Solar particle events eject high-energy protons that may impart a lethal dose of radiation to insufficiently sheltered astronauts.

SP-100 - 100 kWe-class space power system.

Space sub - High-leverage concept developed to assist with on-orbit and planetary assembly and maintenance operations.

Space Transportation System - All hardware systems, and support equipment, facilities, and manpower to deliver payloads to Earth orbit onboard the Space Shuttle.

Space Transportation Integration Agent - The Office of Exploration Space Transportation Integration Agent is responsible for the definition, integration, and operations of all space vehicles beyond low-Earth orbit.

Special Assessment Agent - Directors of independent studies targeted towards the identification of high-leverage technologies, systems, or operational techniques. Special Assessment Agents are truly independent and are not utilized as systems or subsystem definition agents for system designers.

Specific impulse - A performance parameter of a rocket engine, expressed in seconds, equal to the thrust in pounds divided by the weight flow rate in pounds per second.

Specific mass - Ratio of electric propulsion spacecraft's power and propulsion system mass to initial power available. Specific mass is the inverse of specific power.

Stirling engine - An engine in which work is performed by the expansion of a gas at high temperature; heat for the expansion is supplied through the wall of the piston cylinder.

Study Requirements Document - Publication produced by the Office of Exploration presenting detailed prerequisite requirements for human exploration case studies. This document also sets forth study tasks and schedules to which OSSA responds in support of the Office of Exploration overall study effort.

Sublimator - Device used for the heating of solids (usually under vacuum) to the temperature at which the solid sublimes.

Subsystems - Within an element's particular system, pertains to those components which, when integrated together, form the functional hardware and software infrastructure of a system. Examples include power distribution, heat rejection, and data packeting subsystem.

Systems - Within an element, pertains to individual technical discipline areas which, when integrated together, form the functional hardware and software infrastructure of an element. Examples include power, thermal, and data management systems.

"______ System" - Pertains to one or more functionally independent or functionally interdependent elements within a scenario domain, the complement of which must operate in an integrated manner to achieve a major scenario objective. Examples include Earth-to-orbit transportation systems such as the National Space Transportation System or a class of Expendable Launch Vehicles, orbital transfer systems, such as the Orbital Maneuvering Vehicle and Orbital Transfer Vehicle, low-Earth orbit servicing systems such as the international space station, and planetary transportation systems such as the Lunar and Mars Rover. This term is usually used in conjunction with a word-set; e.g., National Space Transportation System, Advanced Launch System.

Technology Development - The development and demonstration of a hardware, software, or human capability with a performance beyond the current state-of-the-art. Examples of technology development in the Exploration Program are: space nuclear power generation (SP-100), automated rendezvous and docking, and medical health care for long-duration missions.

Telerobotic - Referring to automated systems operated remotely.

Torque equilibrium angle - Measurement of stability of a vehicle (e.g., Space Station Freedom) relative to a reference orientation such as local vertical/local horizontal.

Trans-Earth injection - Mars orbital escape and trans-Earth.

Trans-Earth injection system - Propulsion and guidance system for trans-Earth injection.

Trans-Mars injection - Earth orbital escape and trans-Mars.

Umbra - That portion of a shadow which is screened from light rays emanating from any part of an extended source such as the sun.

User - Any organization, group or individual who uses or plans to use the spacecraft, space elements, or space environs associated with the execution of an exploration initiative for scientific, technology development, or commercial application objectives.

Utilization phase - Third phase of the Lunar Evolution and Mars Evolution case studies. The objectives of this phase include:

1. Establish outpost permanence with expanded surface facilities, greater use of Mars resources, and advanced propulsion technologies.
2. Demonstrate global exploration capability.

Wet mass - Vehicle mass including liquids (e.g., propellants) normally present at the beginning of operation.

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16. Abstract

The Office of Exploration (OEXP) at NASA Headquarters has been tasked with defining and recommending alternatives for an early 1990's national decision on a focused program of human exploration of the solar system. The Mission Analysis and System Engineering (MASE) group, which is managed by the Exploration Studies Office at the Lyndon B. Johnson Space Center, is responsible for coordinating the technical studies necessary for accomplishing such a task. This technical report, produced by the MASE, describes the process that has been developed in a "case study" approach. The three case studies that were developed in FY 1989 include:
1. Lunar Evolution Case Study, 2. Mars Evolution Case Study, 3. Mars Expedition Case Study. The final outcome of this effort is a set of programmatic and technical conclusions and recommendations for the following year's work.

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